AIRCRAFT DATA ACQUISITION SYSTEM
FOR
ACADEMIC FLIGHT EVALUATION

By

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United States
Naval Postgraduate School

THESIS

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ABSTRACT

Various methods for investigating the stability and control characteristics of the US2A were considered in obtaining a system that would provide a proper degree of data accuracy, data availability and system reliability yet still be instructional and functional. To this end, a photo-panel system with its various input systems was designed and incorporated into the Aeronautics Department's US2A, BUNO 136533. Installation and component check-out of this photo-panel system was achieved at the Naval Postgraduate School during the period of July 1970 to February 1971. Stability and control flight evaluation utilizing the system was not completed, however, due to an aircraft accident involving US2A BUNO 136533.

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ACKNOWLEDGEMENTS

The authors wish to thank Associate Professor Donald Layton for his continued guidance and assistance throughout the development of this thesis project.

Production, installation and operation of the many sensor systems employed in aircraft BUNO 136533 were achieved through the tireless effort of the technicians in the Aeronautics Department under the direction of Mr. Ted Dunton and Mr. Bob Besel, the Naval Auxiliary Field Maintenance Department, the Electrical Engineering Department Laboratory Staff under Mr. Al White, the Flight Instrumentation Branch of the NASA Ames Research Center, Mr. Don Schlosser of the Test Pilot School Instrumentation Branch, NATC Patuxent River, Maryland, and Miss D. E. Ward, supervisor of the NPGS Photo Division.

I. INTRODUCTION

The support of Naval Aviation requires continuous evaluation of new and operational aircraft and systems. In its support of this requirement the Naval Air Systems Command needs officers well versed in the field of evaluation of flying qualities. To help fill this need the Department of Aeronautics, Naval Postgraduate School, offers a two-course sequence in Flight Evaluation. A necessary portion of this course is the flight laboratory which entails in-flight testing and data collection. Actual flight information can also provide real-time data for aeronautically oriented computer courses and research projects.

Evaluation of flying qualities requires aircraft instrumentation in excess of that normally included in operational aircraft. The rapid data rate provided by flight test maneuvers necessitates an in-flight data system. The aircraft, provided by the Navy for use by the Naval Postgraduate Students, are not equipped for flight evaluation.

To fulfill the needs of the flight evaluation course the Department of Aeronautics requested approval to instrument a Navy US2A for flight evaluation. Permission was granted by Commander, Naval Air Force Pacific, and the aircraft assigned was Bureau Number 136533.

Several instrumentation systems and data acquisition systems were investigated. Based on this research, appropriate systems for evaluating flight qualities and recording the resulting data were installed and evaluated. This installation was accomplished at the Naval Postgraduate School, Monterey, California during the period July 1970 through March 1971.

II. REQUIREMENTS

The basic requirement was to design, engineer, install and evaluate a data acquisition system capable of providing sufficient information to evaluate the flight qualities of the assigned aircraft. In the design of a complete system each component was evaluated with respect to several specific parameters as well as those of a more general nature which affect the entire system. The specific requirements, peculiar to each sub-system, will be discussed in their respective sections.

There are several criteria which are normally considered prior to finalizing the systems design. These are summarized as follows:

1) Number of required inputs.
2) Desired system complexity.
3) Form of final data output.
4) Desired accuracy.
5) Dynamic response
6) Availability of parts, facilities, and finances.

The applicability of these criteria to the specific problem of providing an aircraft data acquisition system for use in an academic environment is discussed in the remainder of this section.

The primary objective in developing a system for student use was to provide the student with a facility which was easy to use and one whose output was meaningful. The output must fulfill the data requirements of the flight evaluation being conducted and the reproduction of the data should facilitate the data reduction burden. The basic inputs

required for academic flying qualities evaluation are control position and force information and those parameters describing aircraft flight conditions. The latter consist primarily of aircraft yaw, pitch and roll angles, airspeed, altitude and normal acceleration.

To provide accurate input data the flight maneuvers required for investigation of various flying qualities must be performed as professionally as possible. Therefore, the operation of the data acquisition system should in no way inhibit the flying capability of the pilot. In multipilot aircraft this requirement is not as binding as in single piloted aircraft but must still be a consideration. The intent of the system is not to teach the pilot how to operate an experimentation network but to produce data for an educational or investigative flight evaluation. Therefore, the system should be designed for simple and comparatively error-free use. For similiar reasons, the preflight and calibration of the system must be short and uncomplicated.

It was necessary to construct the system at as low a cost as might be possible without degrading quality. The most obvious means of lowering cost was to use equipment available at the Postgraduate School and other government facilities. This requirement was coupled with the need for easy and low cost up-keep.

The Naval Postgraduate School is dependent on NALF, Monterey, for assignment of aircraft. Anticipating the probability of switching this system to another type aircraft, it was developed with adaptability as a primary requirement. The basic needs of an instrumentation system adaptable to different type aircraft are:

1) Conformity with flight systems common to most Navy aircraft.

2) Light weight.

3) Small size.

4) Adaptability to electrical systems found in most Navy aircraft.

The final consideration was to anticipate the possible extension of this system into more complex levels of data recovery. Use of the system for research purposes could require greater input capability. Therefore, the system must be compatible with conceivable additions.

III. INSTRUMENTATION

A. PHOTO-PANEL

Several methods of recording data were investigated, including magnetic tape recorders, oscillographs, telemetry and photo-panels. All but the photo-panel are rather expensive. A previous investigation into the use of a magnetic tape recorder at the Naval Postgraduate School disclosed some inefficiencies due to the lack of an in-flight method for checking proper functioning of the equipment and complexity of system maintenance and operation. All of the above methods, except the photo-panel, provide recorded outputs which must be converted to useable information. The real-time output of the photo-panel provides for a reproduction of flight conditions on instruments familiar to the aviator student. Reducing data is no more difficult than reading instruments. This leaves the student more data analyzing time and less data reduction time.

The accuracy of the photo-panel output depends entirely on the accuracy of the instruments. Since the ability to design and produce instrument faces existed locally, the accuracy of the instruments could be designed into the system. The instruments can easily be replaced and a recording system for several different tests is therefore available. Having weighed the advantages and disadvantages of the various types of recording systems, it was decided to adapt the photo-panel for use in this data acquisition system.

The photo-panel concept employs a completely enclosed theatre to capture all internal light and blank out external light. On one end of

the theatre all the instruments are secured to a curved panel. The curved panel provides for a camera-to-instrument distance which is equal for all instruments, thus solving the focus problem. The interior lighting must provide enough light for proper exposure yet be shielded to prevent glare or shadows. The camera is mounted external to the theatre and has an unbroken view of the instrument panel through a hole in the opposite side of the theatre.

A 16-hole theatre frame, which was obtained from the Special Flight Test Instrumentation Pool, NATC Patuxent River, was made available for use by the school (Figure 4). Lighting for this theatre was provided by four 28 VDC, GE 307 bulbs each of which draws 0.625 amperes and produces 20 candle power.

The photo-panel camera, a CIA automax, belongs to the Naval Post-graduate School Aeronautics Department. This 35mm camera is specially designed for photo-panel work with its rugged construction and exceptional close-up ability. The entire camera system consists of the camera, intervalometer and control box used for remote control of the camera by the pilot. The intervalometer provides 28 VDC pulses to operate the camera shutter, gear motor, and take up motor. The intervalometer may be controlled to provide camera speeds of from one-half to eight frames per second. Camera lens settings for this photo-panel, with the existing lighting, were found to be five feet on the distance scale with an aperture setting of f5.6. The film used for this purpose was Kodak Linagraph Shellburst, Gray base Film.

The photo-panel was secured to a shock mounted frame in the forward part of the aircraft's aft compartment (Figure 5). The camera system was

connected to the control box located on the center plate between the pilot's and copilot's instrument panel (Figure 24). This permitted the pilot or copilot to control the photo-panel camera in flight.

The number of system outputs was fixed at 12, leaving the photo-panel's extra four holes for future system expansion. The twelve outputs to be recorded were:

1)	Yaw Angle	7)	Aileron Deflection
2)	Pitch Angle	8)	Rudder Deflection
3)	Airspeed	9)	Elevator Force
4)	Altitude	10)	Aileron Force
5)	Time	11)	Rudder Force
6)	Elevator deflection	12)	Accelerometer

All these except airspeed, altimeter and clock provided d.c. outputs and were best represented using a zero-center position gage. The three-inch hole in the photo-panel fixed the maximum instrument diameter. Three-inch gages photograph well and provide greater accuracy than smaller gages. Eight three-inch d.c. zero-center, ruggedized gages with one-milliamp meter movement were loaned to the Aeronautics Department by NASA Ames. Zero-to-five kilohm variable resistors were connected in series with each gage to provide variable sensitivity to match the output of any system. However, the one-milliamp movement was not sensitive enough for the low power output of the vane system. A 25-microamp movement and one 100-microamp movement gage, both with the same characteristics of the other gages, were obtained for use with the vane system. Zero-to-seventy five kilohm variable resistors were placed in series with each of these gages and the resulting sensitivity was high enough for the vane system, and could be varied to match the pitch or yaw output.

To be photographed and provide an easily readable scale, black faces with white markings were used on all gages. The scale design consisted primarily of determining limits of the outputs to be expected in flight evaluation. These limits were either specified by appropriate instructions or determined from flight experience. Once the limits were defined, scale design was accomplished by transferring the defined scale to a 250° full scale deflection on the instrument face. The linear outputs of all systems made this a simple ratio problem. The designed face was then finished in india ink and converted to a black face with white markings using a reverse negative photographic process. After this face was glued to the instrument metal face plate, final calibration was completed by adjusting gage sensitivity to match gage limits to output limits.

The following sections discuss the various sensors and how their output is prepared for display by the photo-panel instruments.

B. VANE SYSTEM

To provide accurate airspeed, altitude, and yaw and pitch angle indications the sensors must be located clear of the aircraft's effect on local flow. This suggests some means of extending a sensor, capable of measuring the above mentioned flight parameters, into the free stream. The requirements for a sensor used on low performance aircraft are light weight and high sensitivity.

The light weight sensor would cause very little change to the dynamic characteristics of the boom which might be used to extend the flow direction and speed sensors into free stream. The high sensitivity requirement is due to the lack of high dynamic pressures at low speeds.

Two systems presently used are vane and aerodynamic types. Aerodynamic types measure pressure differences and may thus determine direction of flow. Vane types consist of two vanes, mounted on some aerodynamic shape, indicating yaw and pitch as they align with free stream. The latter system is used more often in flight test work as it is inexpensive to produce, easy to use and calibrate. A vane system (Figure 10) was obtained on loan from NASA Ames. This assembly, including a kiel-type pitot, provides for a compact, light weight, flight condition sensor. The kiel tube allows for wide yaw airspeed measurement with negligible flow inclination error. Static ports, located well aft along the tube provide a tangential flow past the ports, thus reducing position error.

The vanes used are very light and therefore sensitive to the low speed flight regime of the US2A. Vane position is sensed by 28 VAC/400 hertz synchro located within the vane assembly. A set-screw located in the shaft connecting each vane with its corresponding synchro provided a means of calibrating vane position and desire output. This output, consisting of an error signal superimposed upon the 28 VAC/400 hertz carrier, was sent to the demodulator for conversion to a DC signal.

The boom, required to extend the vane system into the free stream, should be light weight and of sufficient length. Prior to design of a boom, location must first be determined. The two most often used locations are the wing, and extension from the nose area. Modification of aircraft, required in the latter case, determined the most prudent method to be use of a wing mount. The boom length was defined by the criteria that free-stream conditions near a wing station exist approximately one chord length forward of the leading edge.

To satisfy the low weight requirement while also introducing good machineability, aged 2024 Aluminum Tubing was purchased for the boom construction. Three tubes of different diameter were selected to produce a telescoping structure (see Figure 6). When totally secured, with edges rounded, this gave additional strength and an aerodynamically faired contour. This also provided for an adjustable natural frequency (w_n) by varying stiffness. Tests by Grumman on a similar boom showed that w_n = 10.5 hertz/sec provided a stable boom in all flight regimes common to the US2A. The final dimensional adjustments were made after measuring the natural frequency both in the Naval Postgraduate School subsonic tunnel and on the aircraft during ground operations. To measure this frequency, a Bentley reluctance gage powered by 18-volt power supply was placed adjacent to the boom. Vibrations were induced and the output from the reluctance gage was sent to a counter. The boom was telescoped until an W_n of 10.5 hz/sec was obtained.

The design of the boom mount was predicated on light weight and compatability with the Aero 14A weapons carrier, used on US2A's and US2D's. The carrier was modified and strengthened to support the boom. The bracket was to be semi-permanently installed on the carrier, whereas the boom itself could be easily removed from the bracket. Quick release fittings were provided for the cable and nylon tubing used for signal transmission. The location of the boom and bracket were selected as the outboard station on the starboard wing. This placed the vanes as far away from the propeller disturbance as possible and also provided clearance when folding wings.

The vane and pitot static signals were transferred to the panel through the boom, wing and engine nacelle using shielded cable and nylon tubing.

This tubing provided easy installation and produces no chafing problems with other electrical systems in the wing. The tubing's flexibility also permitted its use in routing through the wing-fold area.

To properly adapt to the DC instruments, the AC error signal from the pitch and yaw vanes was demodulated and transferred to the gages. The possibility of repeating the signal with a synchro gage was investigated and decided against due to the excess lag time observed in the synchro repeaters. Demodulation, although requiring another "black box", was far more desirable for dynamic testing. The demodulator, located aft in the electronic bay (Figure 3), is powered by 115V/400 hertz aircraft power. This is transformed to 28V/400 hertz to power the vane synchros. The error signal, received from the vane assembly, is simplified to DC output using a chopper circuit shown in Figure 13. As mentioned earlier, the power output is low requiring sensitive gages for display.

The first step in calibration of the pitch and yaw system required determination of instrument scale limits. From previous flight experience in the US2A it was determined that the maximum yaw to be expected was twenty degrees and maximum and minimum pitch levels were $+15^{\circ}$ and -5° respectively. These are slightly excessive but sufficient accuracy is provided and the useable limits are covered. To zero the outputs for yaw, a metal calibration scale was mounted below the yaw vane. Zero voltage for zero yaw was obtained by loosening the vane set screw and adjusting the vane appropriately. The same principle was used on the pitch vane with the exception that zero center on the gage was $+5^{\circ}$ of pitch. The output was then connected to the gages and each position checked against the gage. Any discrepencies were of a sensitivity nature

and these were corrected by adjustment of the resistor in series with the gage. See Appendix B for complete calibration procedures.

C. FLIGHT CONTROL SYSTEM

The control surfaces are positioned by either the pilot or copilot. Each pilot has a yoke for elevator and aileron control, and a rudder pedal for directional control. Each set of controls is mechanically connected to the appropriate control surface.

The lateral control system consists of the aileron and spoilers. A spoiler is linked directly to each aileron on the same wing so that when the aileron is deflected up, the adjacent spoiler is also deflected up. The yoke is connected to the ailerons and spoilers by a series of bell cranks and push rods (see Figure 26).

The longitudinal control system consists of elevators which are connected to the yoke by a series of push rods, bell cranks and cables.

Directional control is obtained by movement of the rudder pedals which are connected to the rudder through a system of push rods and cable (Figure 27).

1. Control Force Measuring System

The most efficient means of determining control force inputs was through the incorporation of strain gages. These must be located so that a true measure of the pilots force inputs to the control system can be obtained under varying loads and environmental conditions.

The rudder force sensor consisted of four BLH SR-4, 120 ohm, FAB-25-12S13 strain gages located on the rudder push rods immediately forward of the rudder pedals (see Figure 19). Two gages were fixed to each rod to produce a double-sensitive temperature-compensating wheatstone

bridge circuit, capable of measuring the difference of forces applied to the rudder pedals. The proper connection of the strain gage leads was accomplished in the balance-box which will be more fully described later. This wheatstone circuit used a five-volt d.c. power source, provided by the battery pack also to be described later.

The measure of aileron and elevator forces was accomplished through use of a special control wheel obtained from NASA Ames (Figure 24). Each force system uses four strain gages of the same type as used in rudder system, mounted on cantilever beams located within the control wheel. The cantilever beams are positioned so that one cantilever moves fore and aft as force is applied to control the elevator, and one moves up and down as force is applied to control the aileron. The strain gages, oriented to form a wheatstone bridge circuit for each force system, sense the cantilever deflections thus producing force signals which are sent to the balance-box for adaption to the force gages. This system is powered in the same manner as the rudder system.

The outputs from all wheatstone bridge circuits measuring force inputs were too small for direct instrument pick-up. The required amplification was provided by a Fairchild integrated amplifier circuit shown in Figure 18. Some of the advantages obtained with this circuit are:

1) compact (solid state)
2) inexpensive
3) temperature compensating
4) low noise
5) low drift

6) high gain

The amplifier circuits are located in the balance-box, convenient to both the whetstone circuit outputs and the battery pack, source of its ± 15 VDC power.

The initial calibration of this circuit, fully described in Appendix B, was extensive but, once completed, future calibrations should be easy and require little time. Using the balance pot incorporated in the amplifier circuits, with zero amplifier input, the amplifier circuit output was adjusted to zero. Putting the balanced bridge across the input of the amplifier circuit, an output was induced from the amplifier circuit which had to be removed. Input bias, supplied by three 1.3V mercury batteries, was employed to cancel this undesirable output. The three 1.3V power sources, isolated to prevent crossfeed, were separately connected to the amplifier inputs through off-on switches. Potentiometers were incorporated to provide adjustment of the bias. This was the final adjustment to be made in the initial calibration and it is anticipated that future preflight calibrations will simply consist of zeroing the force indications using the wheatstone bridge balance pots located on the balance box.

The several power inputs mentioned in the force system have a common source in the battery pack shown in Figure 17. This pack, using 26 1.3V mercury batteries for power supply, provides ±15 VDC for the amplifier circuits and +5 VDC for the wheatstone bridge circuits. The steady output, long life and compact size of this pack made it an ideal system for adaptation to any aircraft.

Once calibrated, the amplifier outputs were connected to the d.c. gages for data display. The scales, whose limits were specified

by MIL-F-8785 (ASG), are as follows:

1)	Aileron	50 lbs left and right
2)	Elevator	50 lbs pull, 20 lbs push
3)	Rudder	180 lbs left and right

The gage's variable resistors were adjusted to provide proper correlation between actual force applied, measured by carry-aboard force gages, and indicated force. With the connections complete and the calibration satisfactory, the instruments were installed in the photo-panel theatre.

2. Control Surface Position Indicating System

The measurement of control surface position is one of a displacement nature and, as such, lends itself to several methods of attainment. Two of the more common methods are mechanical measurement, with its output transmitted to pointer indication, and electrical sensor, with electrical signal inputs to gages. Mechanical measurement is very accurate; however, the distance for transmission of the measurement must be kept to a minimum, thus severely restricting the location of this type of system. The electrical sensor, converting a displacement to voltage or current, provides a measurement signal easily transmitted over long distances to the display location. The ease of signal transmission, overriding the slight loss in accuracy, led to the selection of electrical sensors for measurement of control surface displacement in this instrumentation system. Within the field of electrical sensors, the potentiometer transducer, with its simple construction, easy operation and adaptability to 28 VDC power systems, is ideal for aircraft use.

The transducer used for the measurement of aileron and rudder deflection is shown in Figure 31. Variation in resistance of this

system is produced when the slide contact (attached to the shaft which is housed in the potentiometer drum) is rotated, making contact with a single turn 5000-ohm potentiometer. The shaft, rotating under tension provided by a spring within the drum, is connected to a spool at its free end. Wire, secured to this spool, is attached to a cable or rod, thereby correlating movement of the spool with movement of the cable or rod. In the case of the aileron, movement of the aileron push rod was linear with respect to aileron control surface movement thus providing a desirable location for transducer measurement of aileron control surface deflection. (See Figure 29.) The rudder deflection transducer was connected to the rudder cable passing through the aft compartment. This location, shown in Figure 30, also provided linear output with rudder deflection.

A 10,000 ohm sliding arm potentiometer was used for measurement of elevator deflection. The potentiometer housing was secured to the elevator bell crank assembly frame, while the potentiometer slide was attached to the elevator push rod as shown in Figure 28.

The three surface-deflection transducers are powered by a 5 VDC converter which uses 28 VDC/400 hertz aircraft power. Using the circuit shown in Figure 25 the five-volt potential is modified to produce ± 1.4 volts at each end of the transducers. This produces a plus-and-minus output corresponding to plus-and-minus control surface deflection as well as a zero output for zero control deflection.

Calibration is initiated by adjusting the two 10,000-ohm potentiometers in series, parallel with the transducers, to obtain the ± 1.4 voltage mentioned above. By locking the control surfaces at zero deflection and positioning the transducer connections to the cables or

push rods for zero transducer output, the proper zero center is obtained. This adjustment should be of a somewhat permanent nature.

Instrument scales for the deflection system, predicated on limits defined by NATOPS, are listed below:

Control Surface	Limits of Deflection
Rudder	21° left 21° right
Aileron	20° left wing up 20° right wing up
Elevator	15° down 25° up

Final adjustments made to the instrument potentiometers matched the system outputs to the corresponding instrument indications.

D. ACCELEROMETER

Aircraft normal acceleration measurement is provided by the Statham Model A5TC-8D-350 accelerometer, shown in Figure 34. Acceleration sustained by this sensor causes deflection of a force-summing member housed within the instrument. Strain gages, positioned between a fixed frame and the moveable force-summing member, form a wheatstone bridge so that the movement of the force-summing member is sensed. The wheatstone bridge balance, thus altered, produces an electrical output linearly dependent on the amount of acceleration.

Bridge output signal amplification is provided by Statham Model CAO-3-12594 strain gage signal amplifier. The complete circuit, shown in Figure 32, provides voltage outputs ranging from -.2 volts to +.9 volts for normal acceleration ranging from -1g to +6g's, respectively, with zero-volt output corresponding to zero g's.

Calibration of the accelerometer was checked using a static calibration technique called the turnover method. Calibration using a centrifuge method is far more extensive and had previously been completed on this system. Static calibration, used to verify the centrifuge calibration, was initiated by adjusting the system output to zero for an accelerometer orientation perpendicular to the earth's gravitational field, simulating zero g. Turning the accelerometer upright to produce one g, the output was adjusted to .187 volts. Flipping the accelerometer inverted, to establish negative one g, the voltage output should be -.187 volts. Results of this calibration are shown in Figure 35.

Scale limits chosen for display of normal acceleration ranged from negative four g's to positive four g's. This adequately covers the NATOPS g limits of negative one g to positive three g's.

E. SUPPORT INSTRUMENTATION

Support instrumentation is that basic flight instrumentation common to a standard, pilot's instrument panel which has been duplicated in the photo-panel due to the requirement of such data in flight evaluation calculations. Three to be mentioned in this category are altimeter, air speed indicator, and timer.

The altimeter used in this system is identical to that used in the pilot's and copilot's instrument displays. It consists of an evacuated diaphram assembly and a means of transmitting diaphram deflection to three pointers. The first pointer indicates hundreds of feet, the second indicated thousands and the third ten thousands. Field barometric pressure value may be set and displayed in the Kollsman window, thus

enabling a gear system in the altimeter to make corrections for differences arising from using a reference sea level pressure other than the standard 29.92 in Hg.

The altimeter was connected to the static line from the vane systems. Calibration runs in the aircraft from sea level to 7,000 ft. showed agreement within ten feet between the photo-panel altimeter and the pilot's and copilot's altimeter systems, which use a separate static source.

Two different methods of indicating air speed were investigated. The differential pressure transducer, which converts a difference of pressure to an electrical signal, offers many advantages including minimal lag time, easy transmission of signal, and small size. However, the overall cost and need of a linearizing circuit to modify the output for adaption to standard d.c. instruments were major considerations which precluded using this system. It was therefore decided to use a readily available Navy pressure airspeed indicator.

The airspeed indicator used, identical to the pilot's and copilot's indicator used in the US2A, contains sensitive pressure units which indicated the difference between static and pitot pressures. The static and pitot pressures, transmitted by nylon tubing from the vane system, are introduced to opposite sides of a diaphram housed in the instrument. Slight movements of the diaphram, caused by changes in either pitot or static pressure, are transmitted to the pointer which indicates corresponding airspeeds on a 0-400-knots scale. The entire system--vane, tubing and indicator--was calibrated in the Naval Postgraduate School subsonic wind tunnel. The apparatus and calibration results are shown in Figure 8 and Figure 16.

A timer is mandatory for correlation of data in dynamic testing. The ideal timer for photo-panel display is a digital indicator, accurate to tenths of seconds, with a remote reset option and a repeater located on the pilot's/copilot's instrument panel. The excessive cost of such a timer and the availability of a standard Navy elapsed time clock resulted in use of the latter. Utilization of the elapsed-time clock in this system requires a time check prior to flight in order to synchronize the clock of the photo-panel with the clock in the cockpit.

Included in the theatre's instrument panel is a plexiglass frame used to display a small sign indicating what type of evaluation is being recorded, data, pilot, etc.

IV. CONCLUSIONS AND RECOMMENDATIONS

The instrumentation system which was installed on US2A, BUNO 136533, is capable of providing the data required for academic flight evaluation. All installed sub-systems functioned as anticipated and proved to be easy to calibrate, preflight and operate. Complete system calibration and preflight procedures are presented in Appendices B and C. Loss of the instrumented aircraft, due to an accident on 17 February 1971, precluded application of this system to a flight evaluation. Component design and adaptability does, however, provide for installation of this system in another aircraft.

Instrument data recorded by the photo-panel camera is processed by the Naval Postgraduate School and made available for student viewing within one or two days. Recorded data (see samples in Figure 36) may be viewed on the Aeronautics Department Recorder Film Reader which greatly enlarges the picture, making accurate data reduction an easy task. Smaller photo-panels using mirrors to provide proper focal lengths might be required for smaller aircraft, whereas the present theatre should easily adapt to a US2D or an aircraft of comparable size.

The pitch and yaw sensors of the vane assembly provided a smooth, sensitive and accurate output. However, the airspeed system lag would definitely be excessive in a higher performance aircraft and the employment of a pressure transducer sensor should be considered. Although the boom met all requirements for the US2A, other aircraft

will probably require a different boom. Wing mounting of the boom proved very successful and should be favorably considered as a mounting location for other aircraft.

Surface deflection indication was very satisfactory and the system will require only minor alterations prior to installation in another aircraft. The alignment of the slide potentiometer, used to measure elevator deflection, is very sensitive. The use of a transducer, similiar to those used in the rudder and aileron system, might prove more acceptable. Wires connecting the transducers to the associated control cable had a tendency to slip off the transducer disc. Grooves in the disc should be deepened to prevent this possibility.

The force system, providing accurate outputs, simple calibration and easy operation, should be acceptable for use in any aircraft. Location of strain gage installations will depend on the type of aircraft to be instrumented. The balance-box with its associated power pack as used in this system is ideally suited for providing amplification and calibration of the strain gage outputs. Its compact size should easily adapt to any aircraft.

Purchase of a resettable digital timer should be considered for use in another aircraft. Photographs of the digital timer are easier to read, and provide improved post flight data correlation, when used in conjunction with a pilot's digital repeater.

Even though the necessary data for flight evaluation is available with this system, further additions should be considered. Installation of a position and rate gyro system would provide pitch, roll and yaw rate data thereby greatly expanding the flight evaluation capability of this system. Installation of an outside air temperature gage in the

photo-panel would provide a permanent record of temperature data necessary in many flight evaluation calculations. Qualitative interpretation of aircraft response could be provided by a carry-aboard cassette tape recorder, easily connected to the ICS system. This would correlate well with the quantitative results obtained from the photographic data.

The academic need of this airborne data acquisition system has previously been discussed. Valuable correlation data for ground simulation research, presently being carried on in the Aeronautics Department, could be provided by this system. Development of an in-flight recorder for use in accident investigation is of great interest to the Navy. Research oriented towards this objective would provide another effective use for this data acquisition system. Considering the capability of the data acquisition system, the need for the system, and the ease with which the system could be installed in another aircraft, it is highly recommended that the Aeronautics Department obtain another aircraft and that the system be installed with the noted improvements.

APPENDIX A

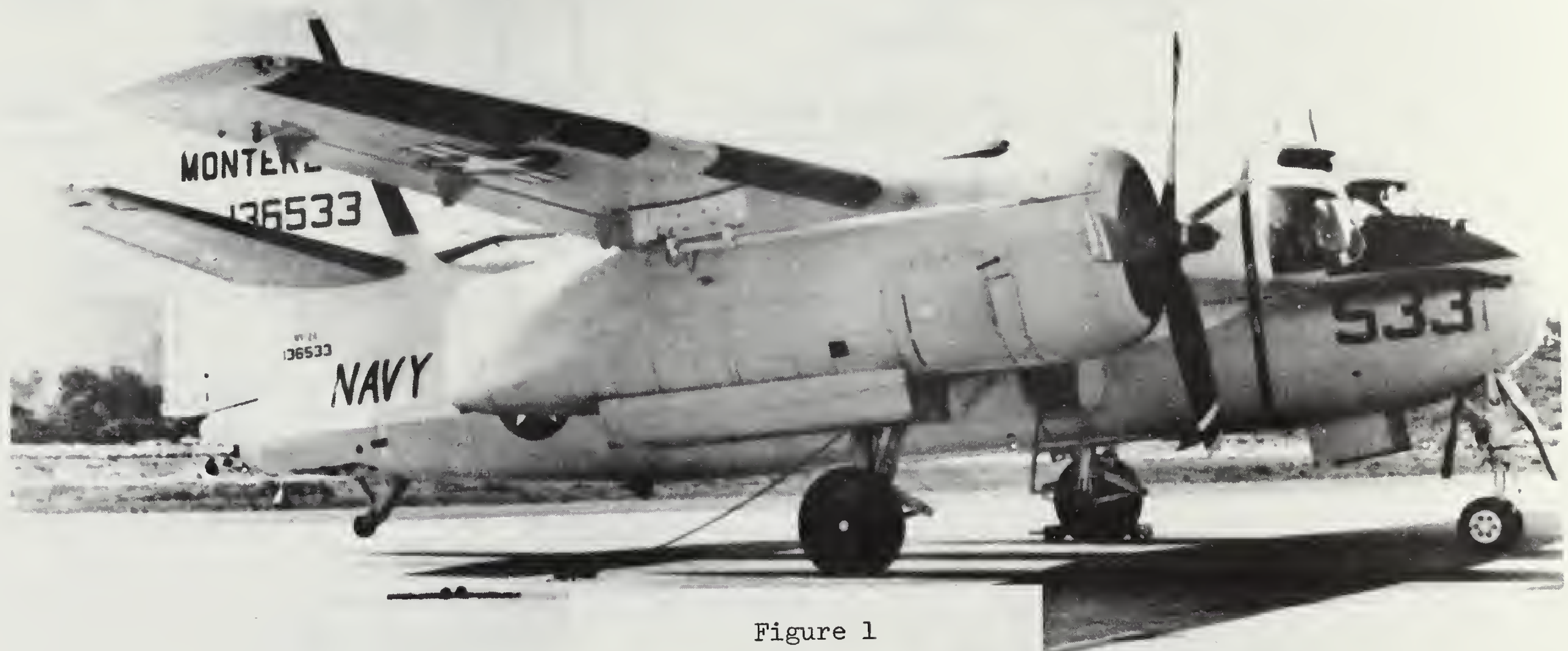


Figure 1

US2A Aircraft BUNO 136533

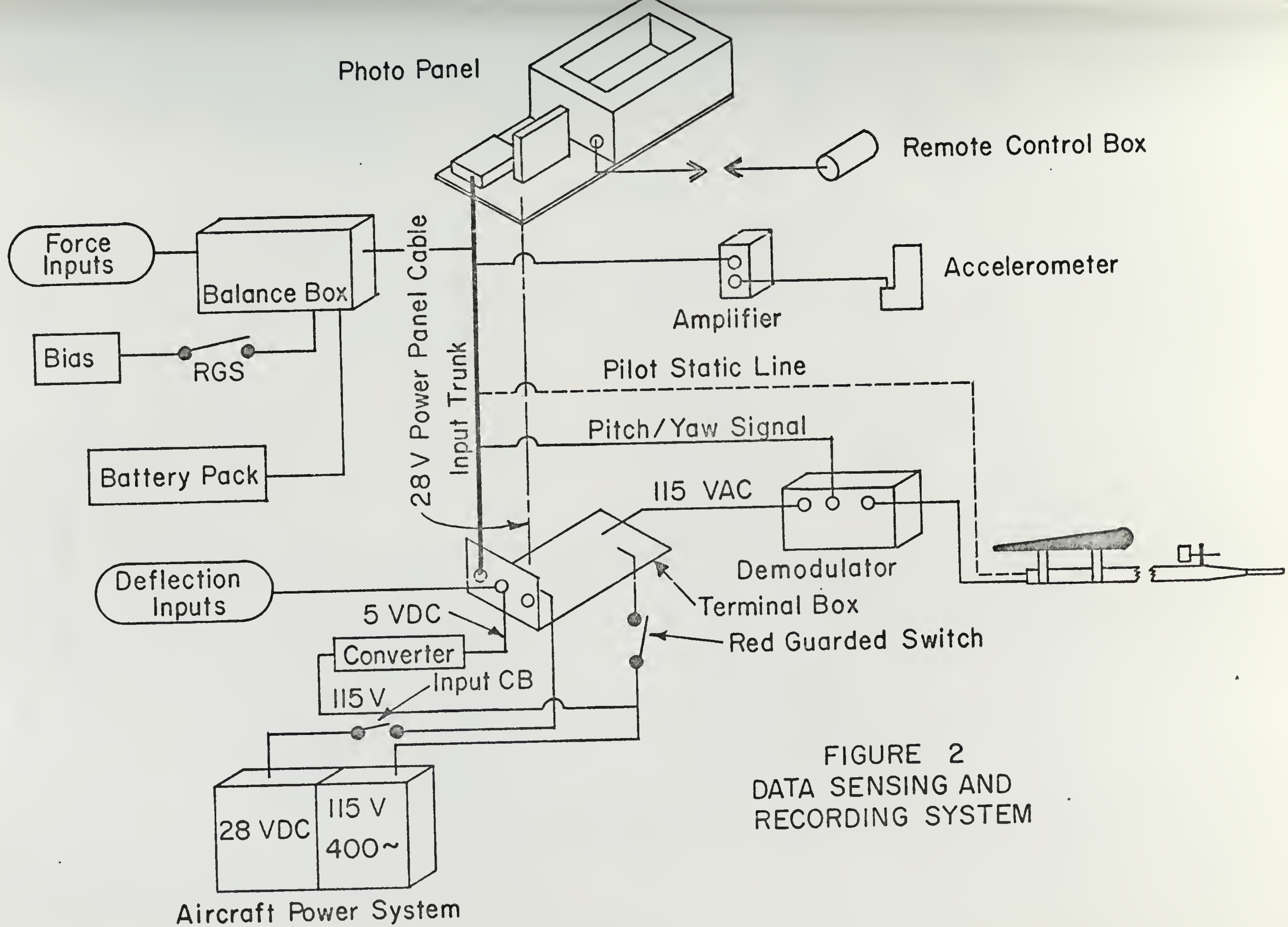
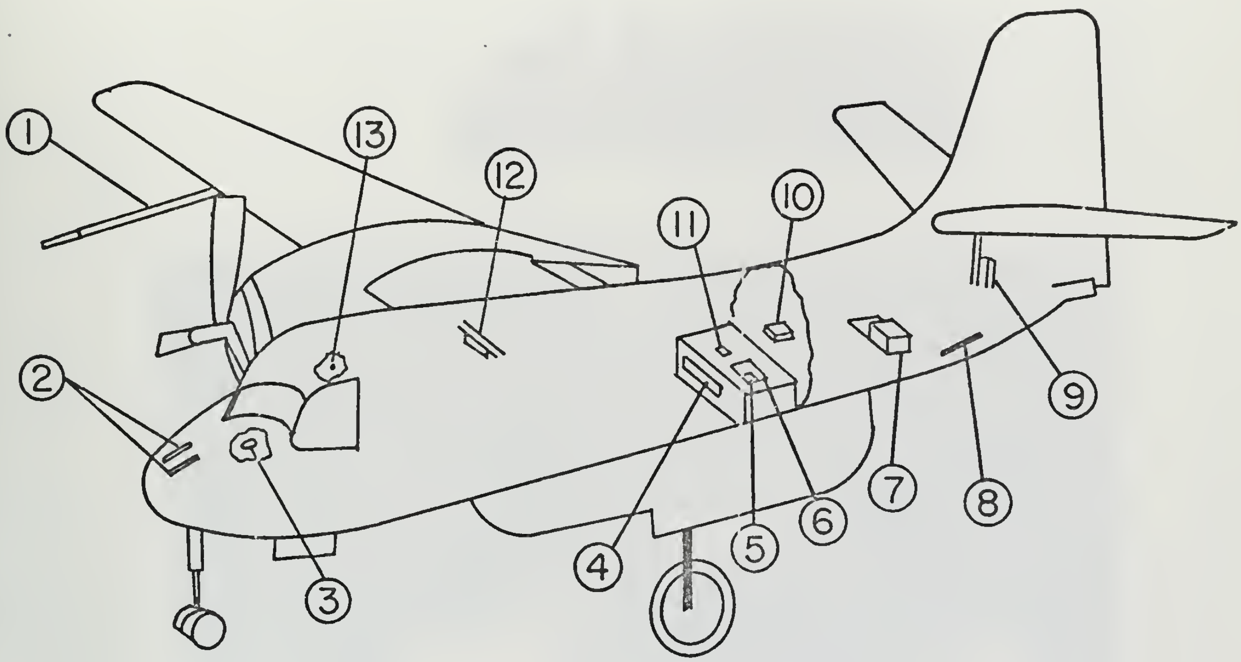


FIGURE 2
DATA SENSING AND
RECORDING SYSTEM



1. Flight Boom	8. Rudder Transducer
2. Rudder Rods	9. Elevator Transducer
3. Control Panel	10. Demodulator
4. Battery Pack	11. Terminal Panel
5. Bias Box	12. Aileron Transducer
6. Balance Box	13. Photo Circuit Breaker
7. Photo Panel	

FIGURE 3
COMPLETE DATA SENSING AND
RECORDING SYSTEM INSTALLATION

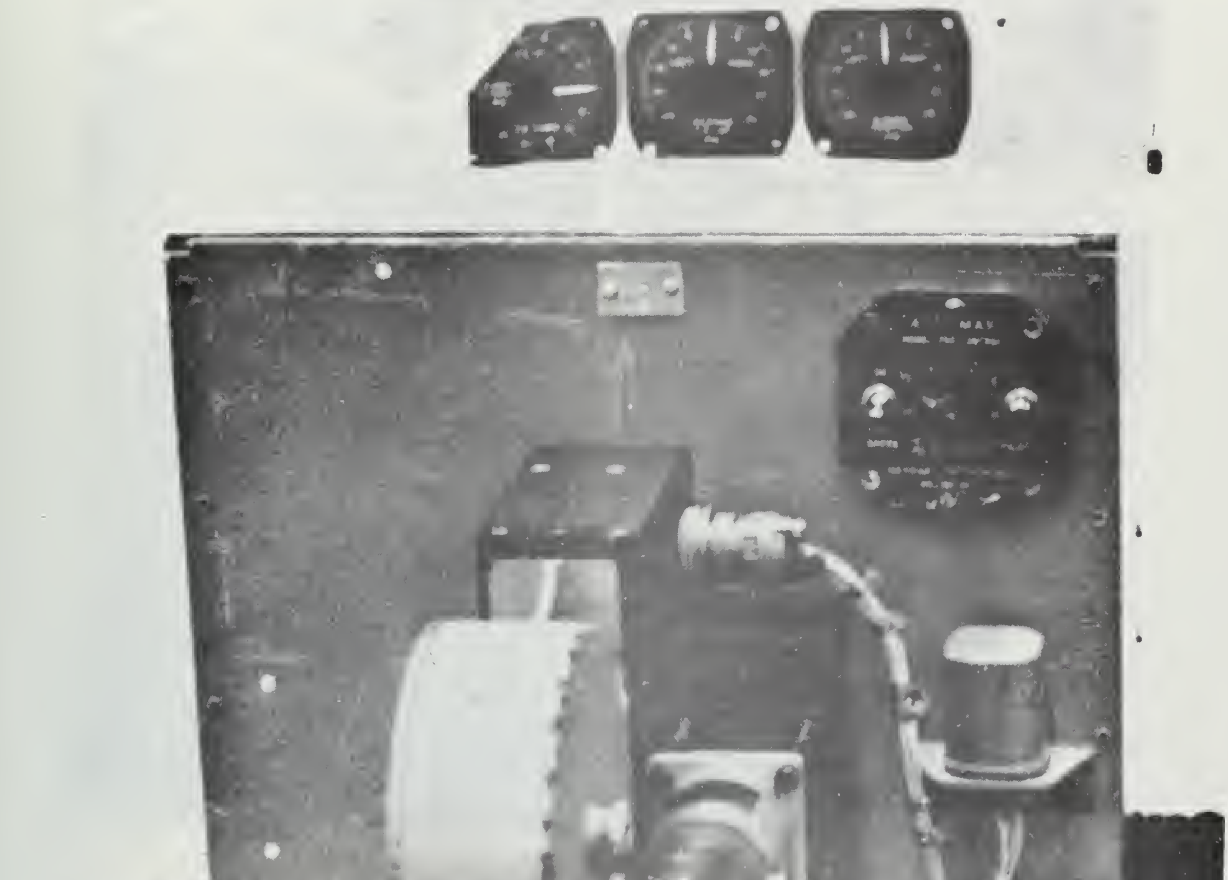


Figure 4

PHOTO-PANEL SYSTEM

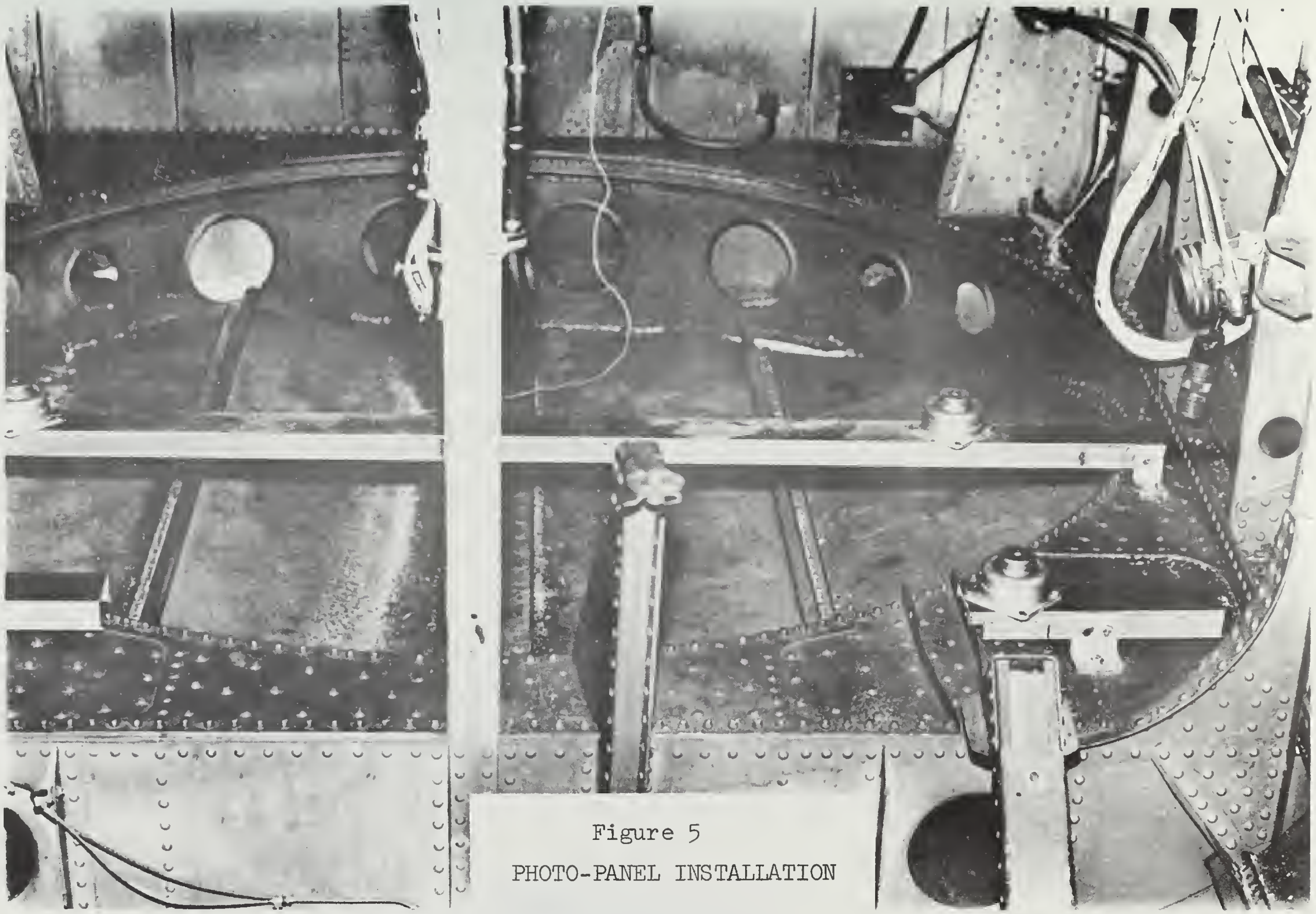


Figure 5
PHOTO-PANEL INSTALLATION

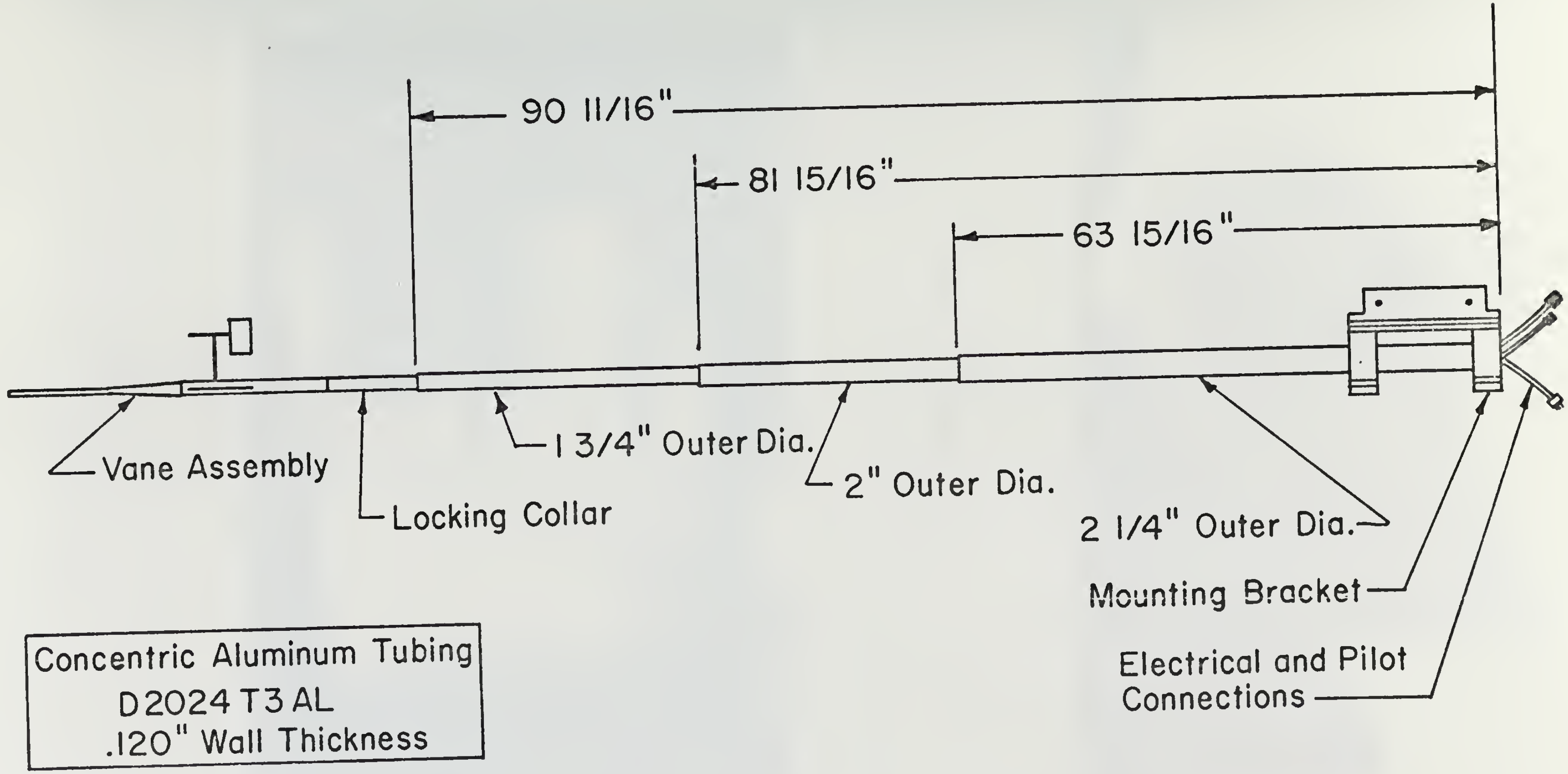


FIGURE 6
FLIGHT BOOM ASSEMBLY

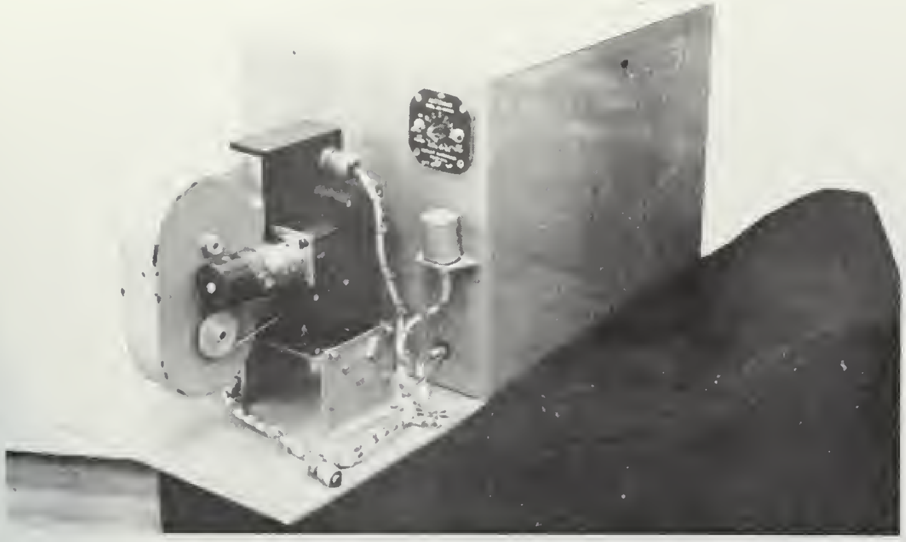


Figure 7

AUTOMAX CAMERA SYSTEM

Figure 8

PITOT-STATIC CALIBRATION SET-UP

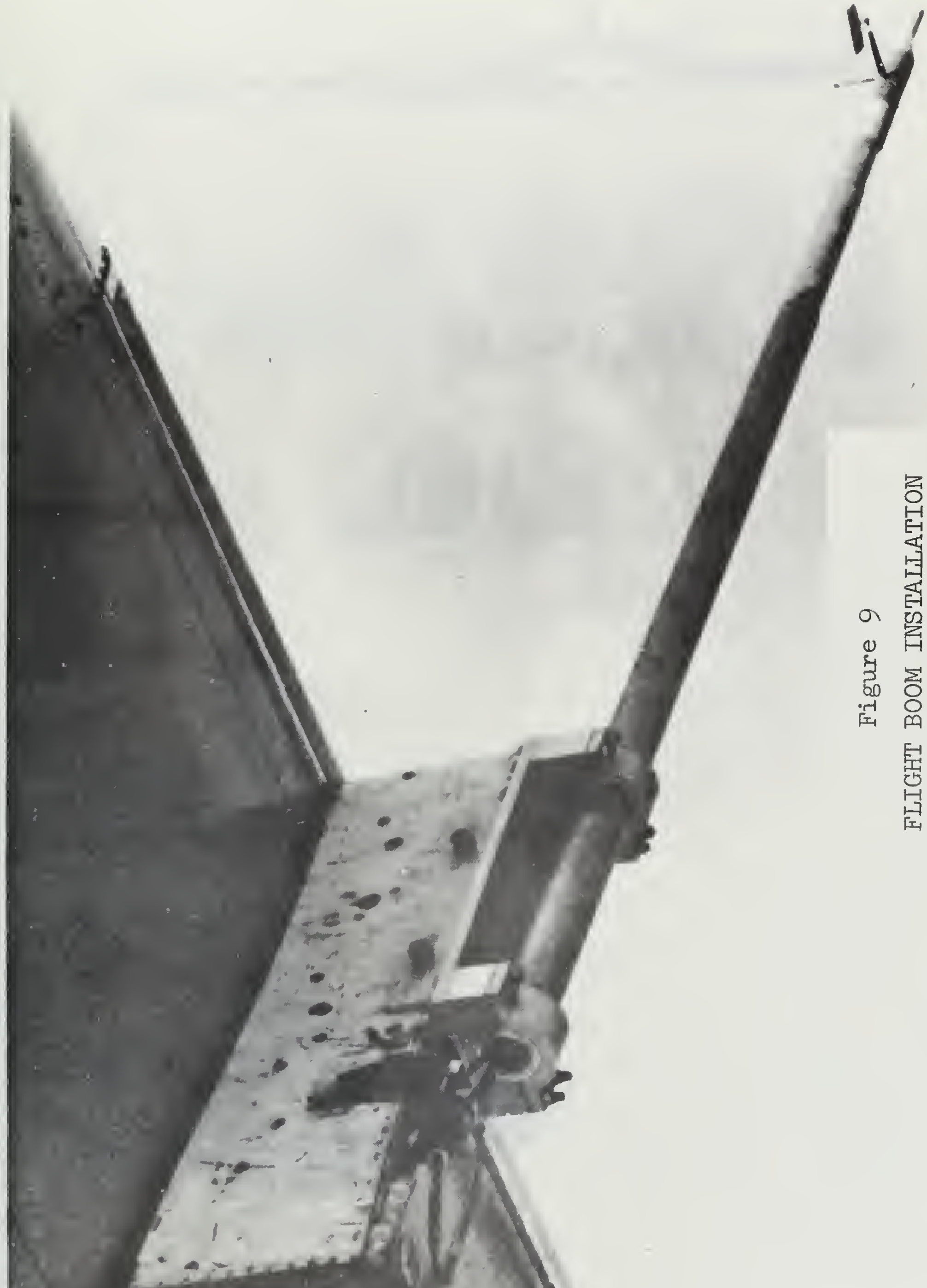


Figure 9
FLIGHT BOOM INSTALLATION

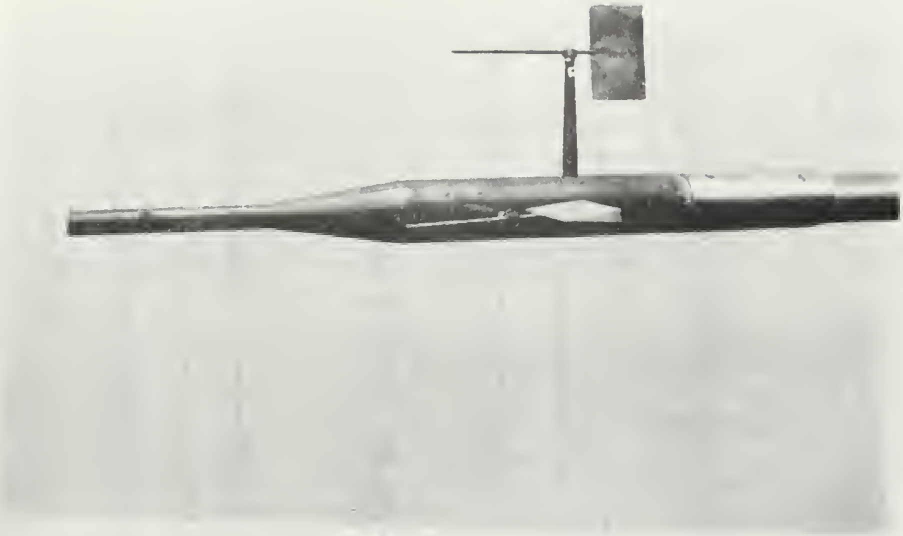


Figure 10
RYAN VANE ASSEMBLY

Figure 11
BOOM SYSTEM IN FLIGHT

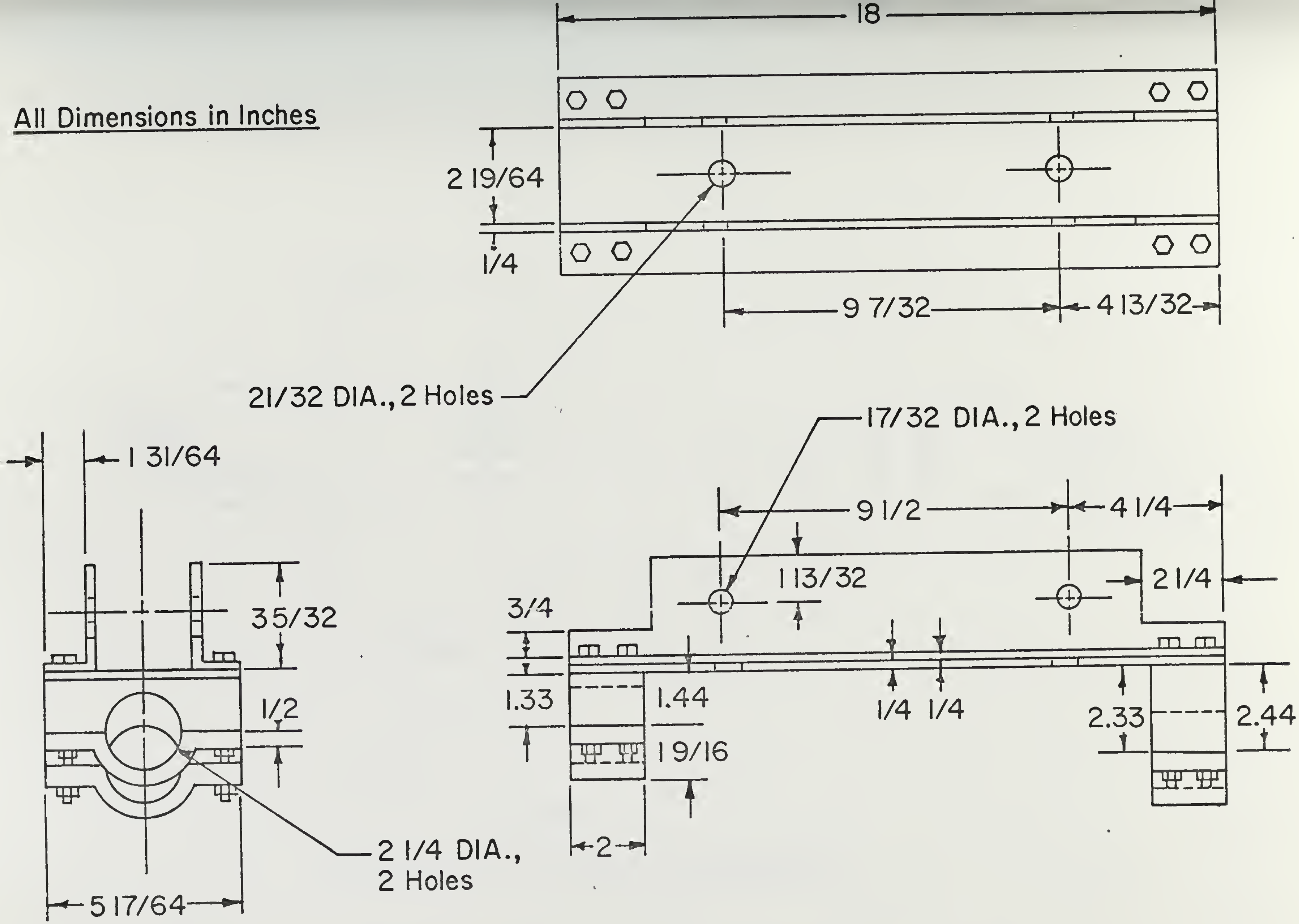


FIGURE 12
FLIGHT BOOM MOUNTING BRACKET

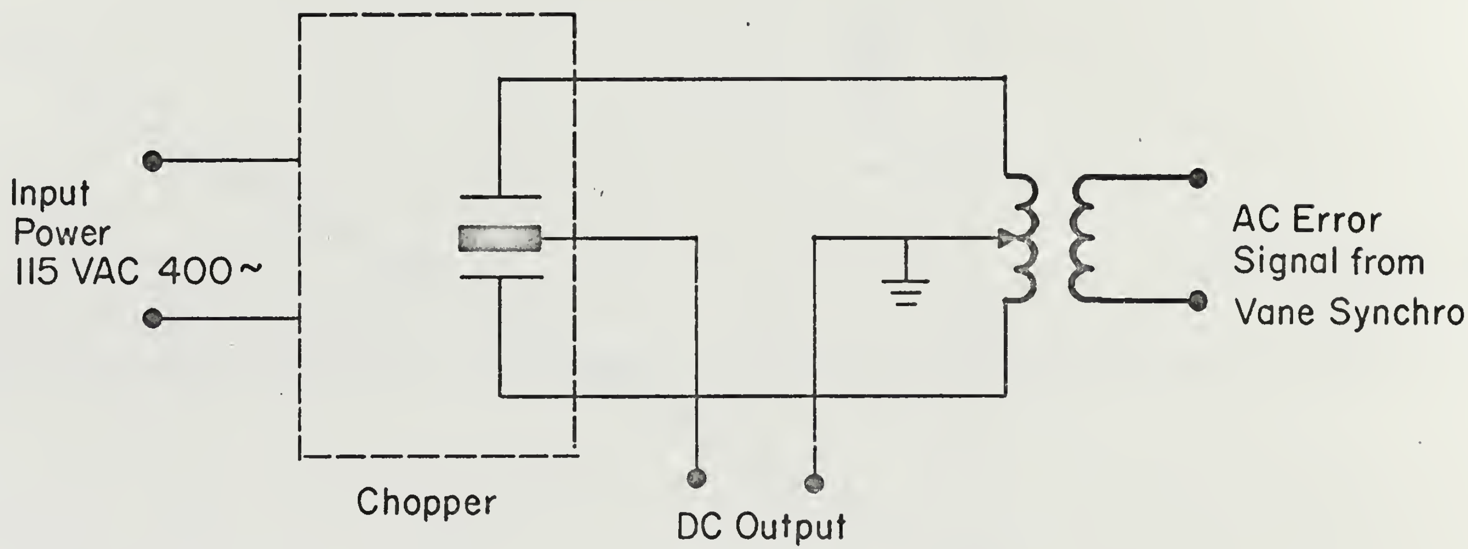


FIGURE 13
DEMODULATOR CIRCUIT SCHEMATIC

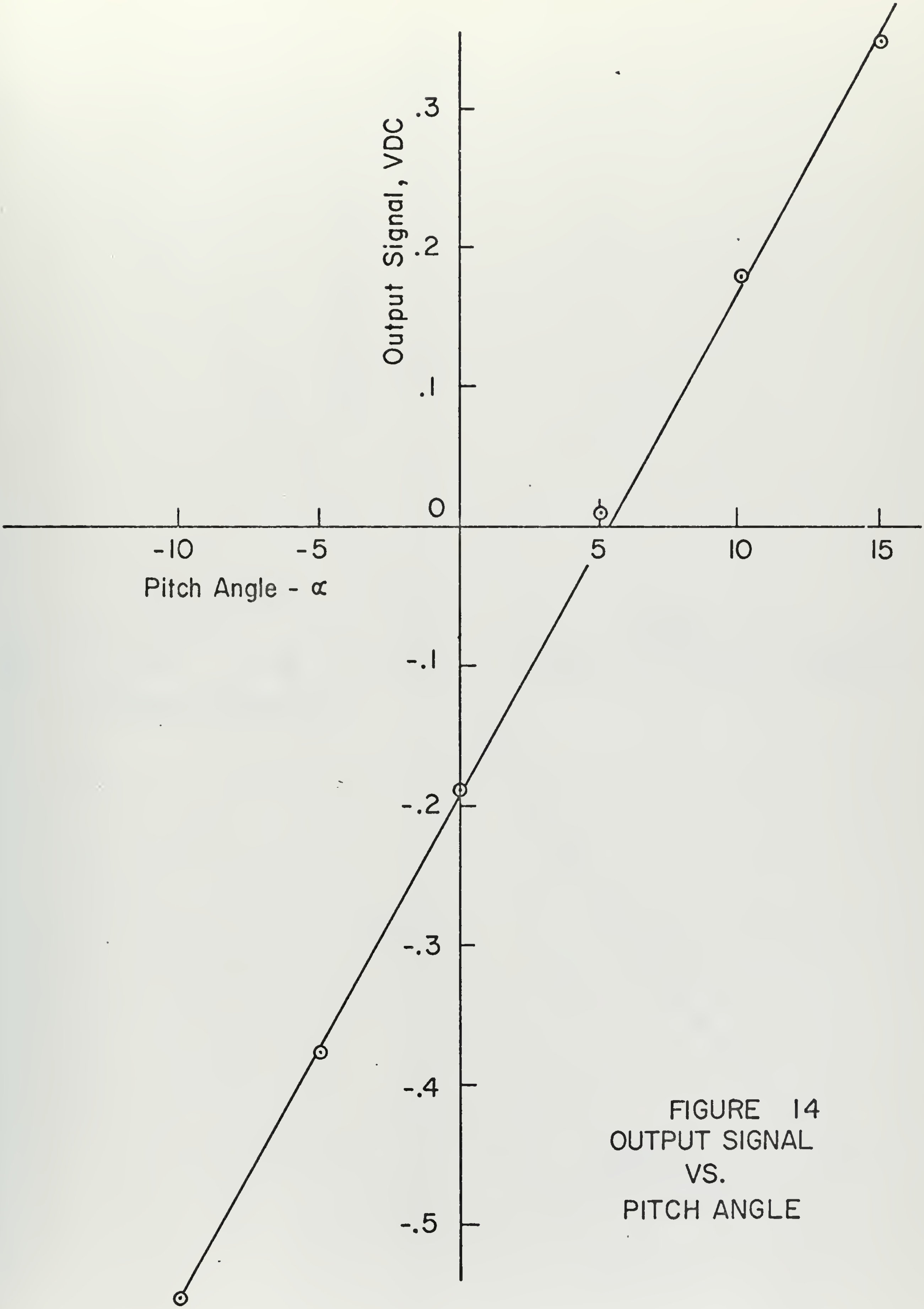


FIGURE 14
OUTPUT SIGNAL
VS.
PITCH ANGLE

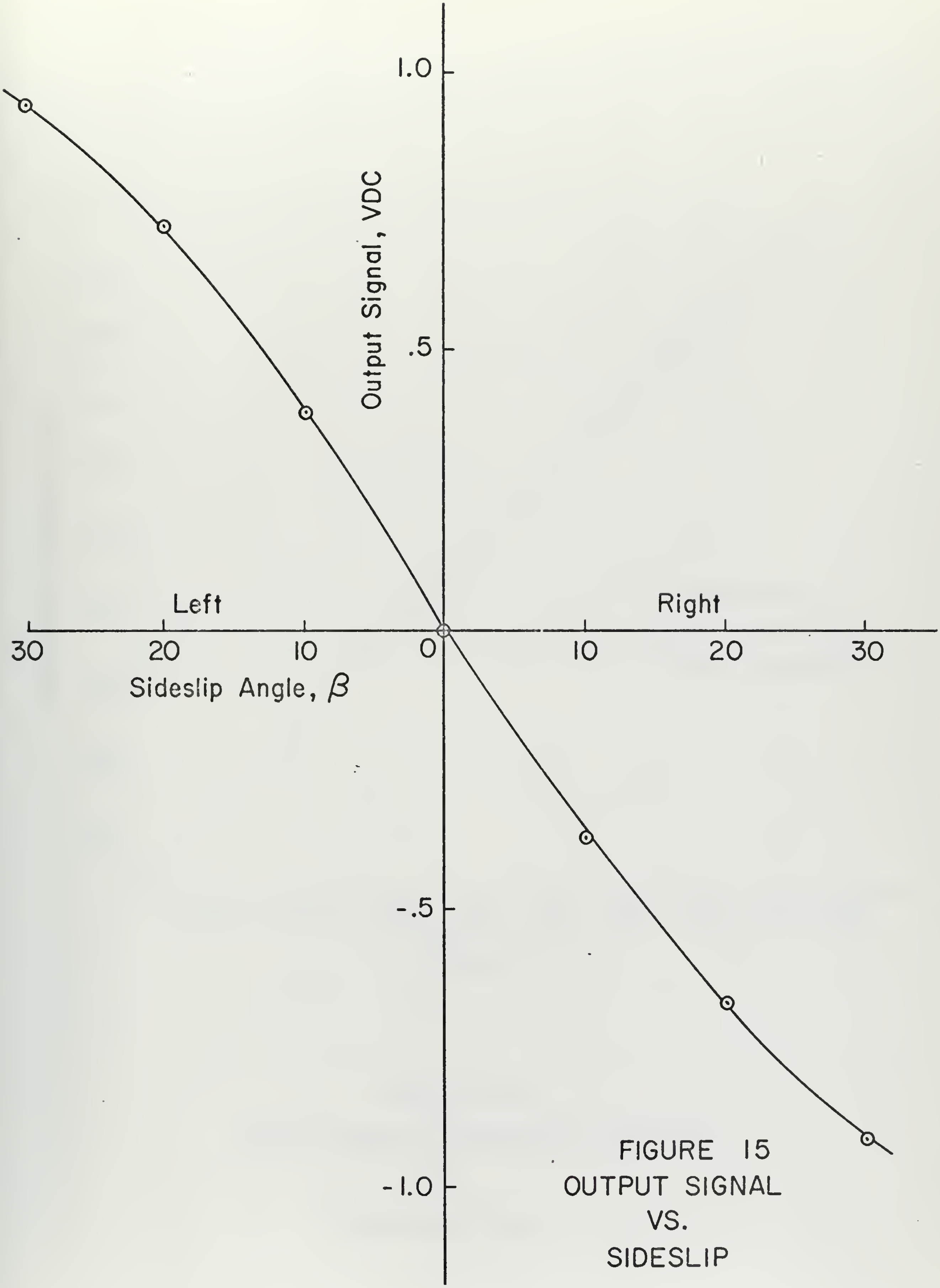


FIGURE 15
OUTPUT SIGNAL
VS.
SIDESLIP

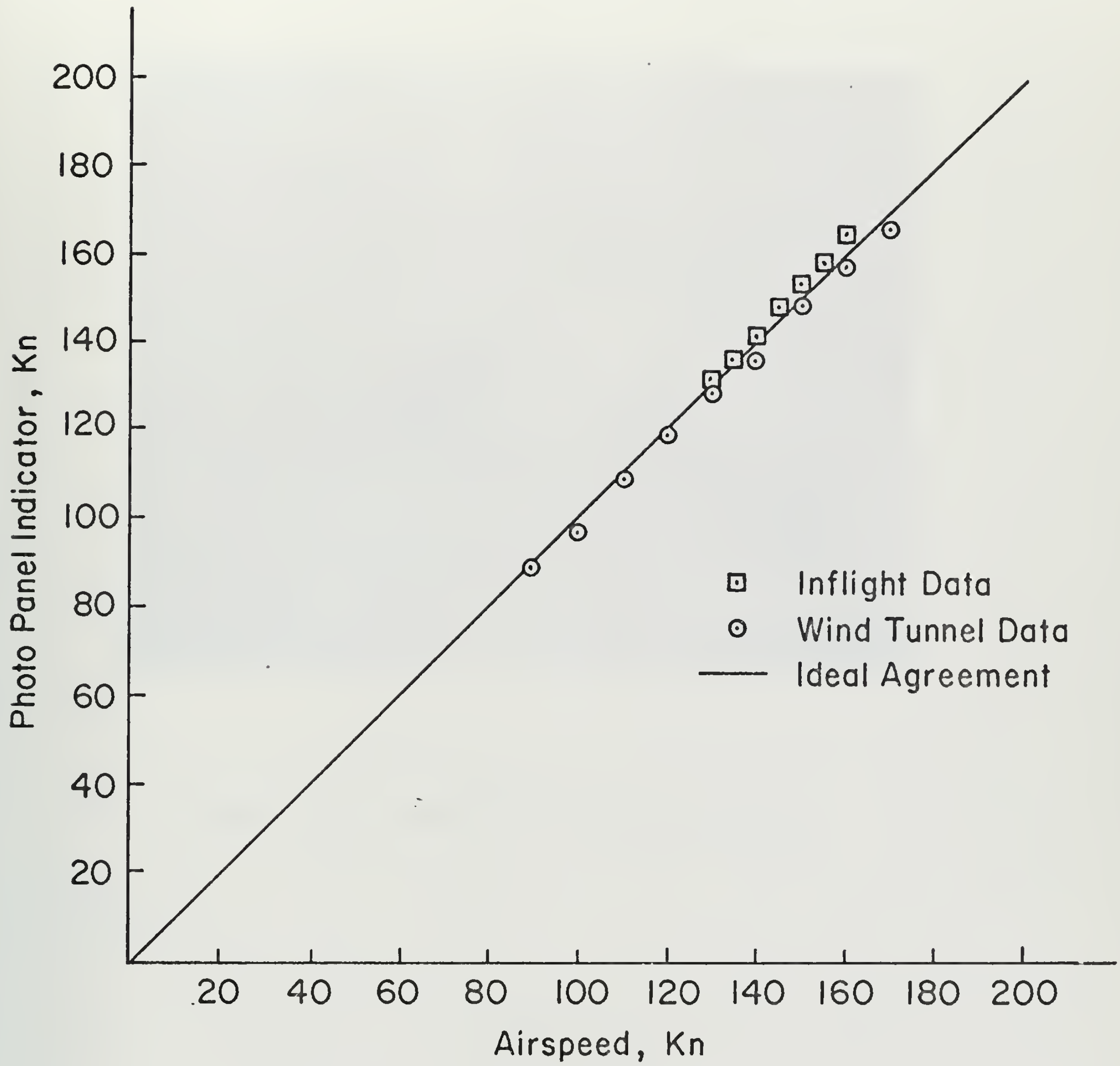


FIGURE 16
PHOTO PANEL INDICATOR AIRSPEED
VS.
REFERENCE IAS

Figure 17

CONTROL FORCE BALANCE AND AMPLIFYING SYSTEM

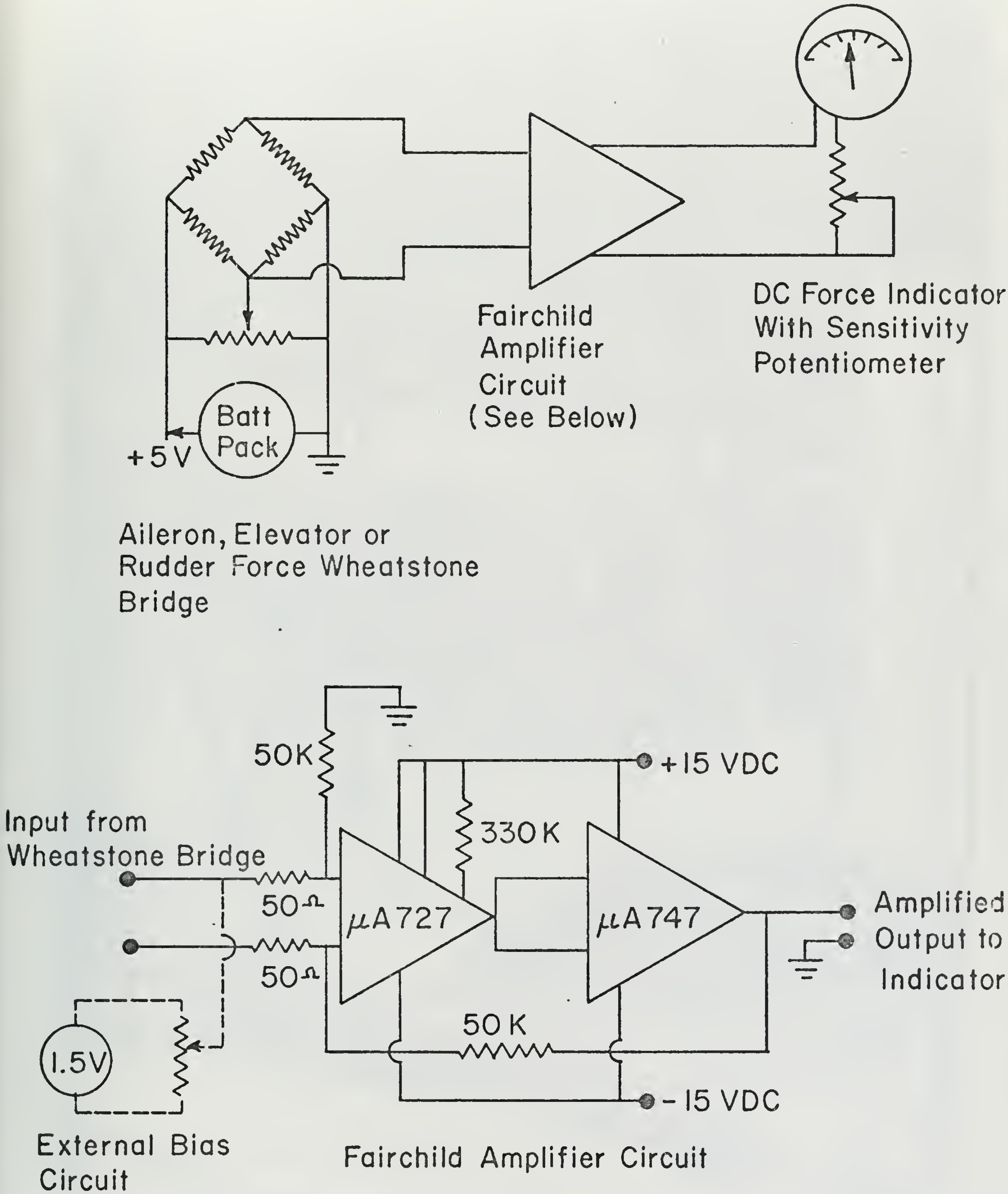


FIGURE 18
FORCE INDICATING SYSTEM

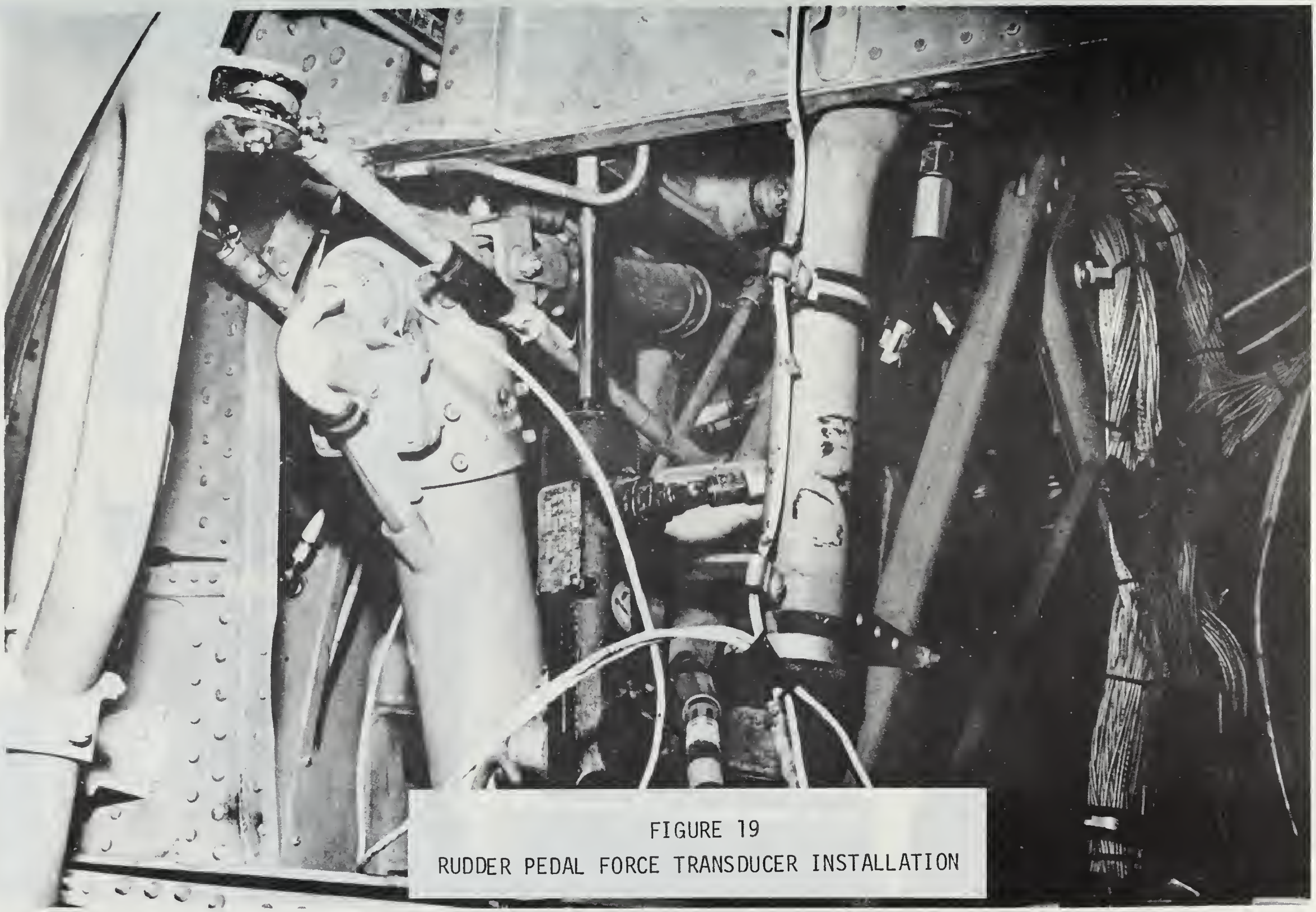


FIGURE 19
RUDDER PEDAL FORCE TRANSDUCER INSTALLATION

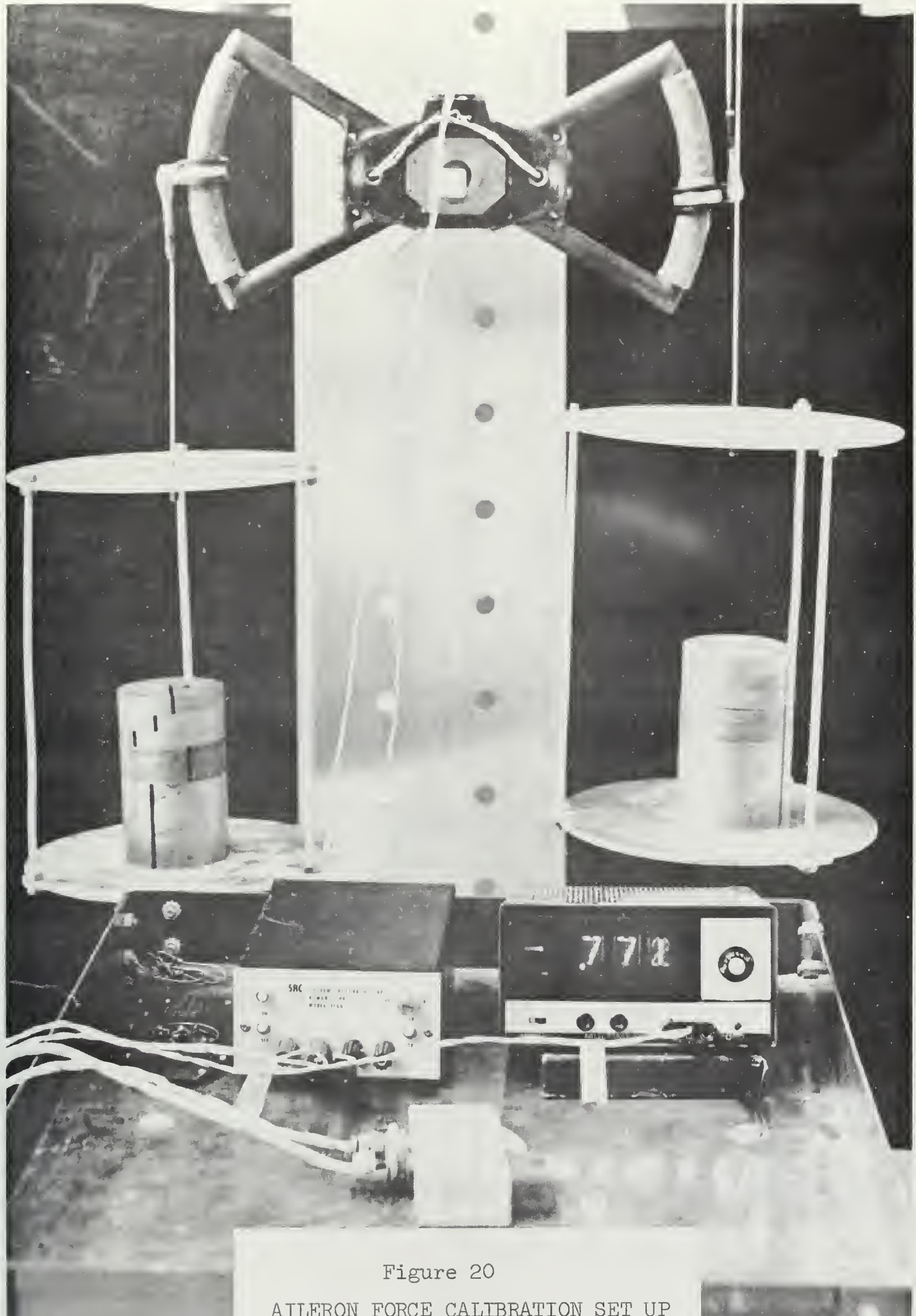


Figure 20

AILERON FORCE CALIBRATION SET UP

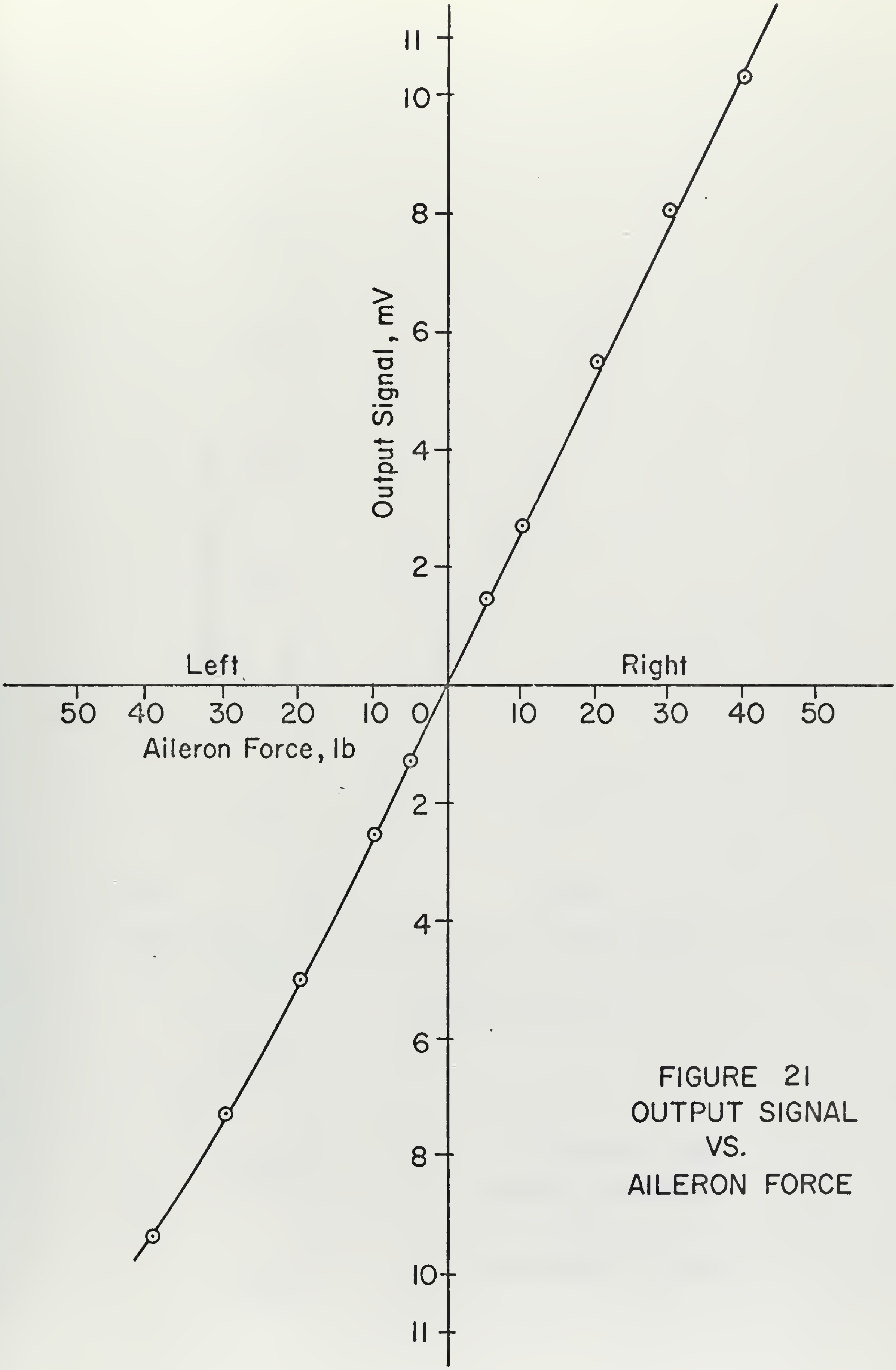


FIGURE 21
OUTPUT SIGNAL
VS.
AILERON FORCE

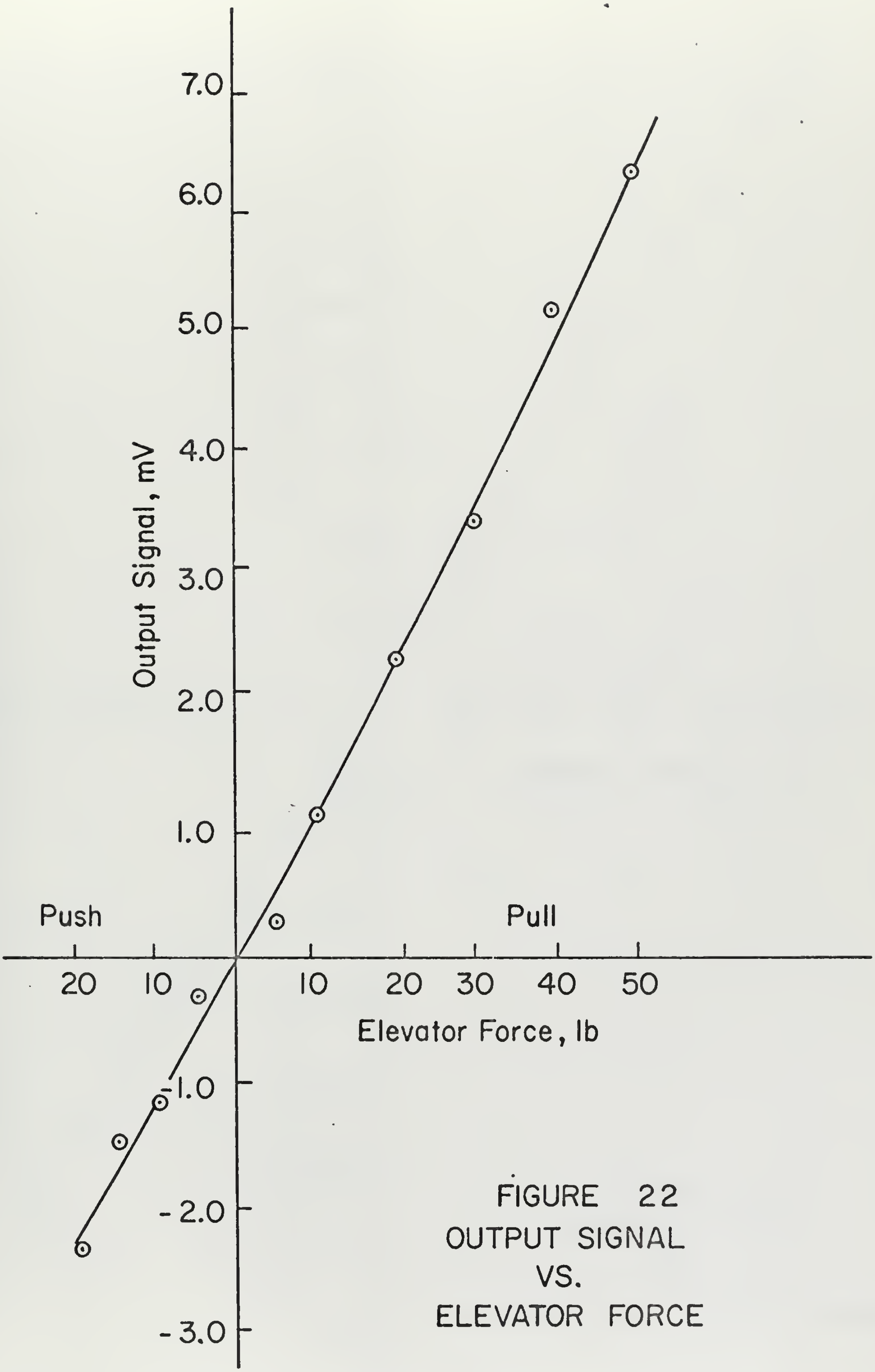


FIGURE 22
OUTPUT SIGNAL
VS.
ELEVATOR FORCE

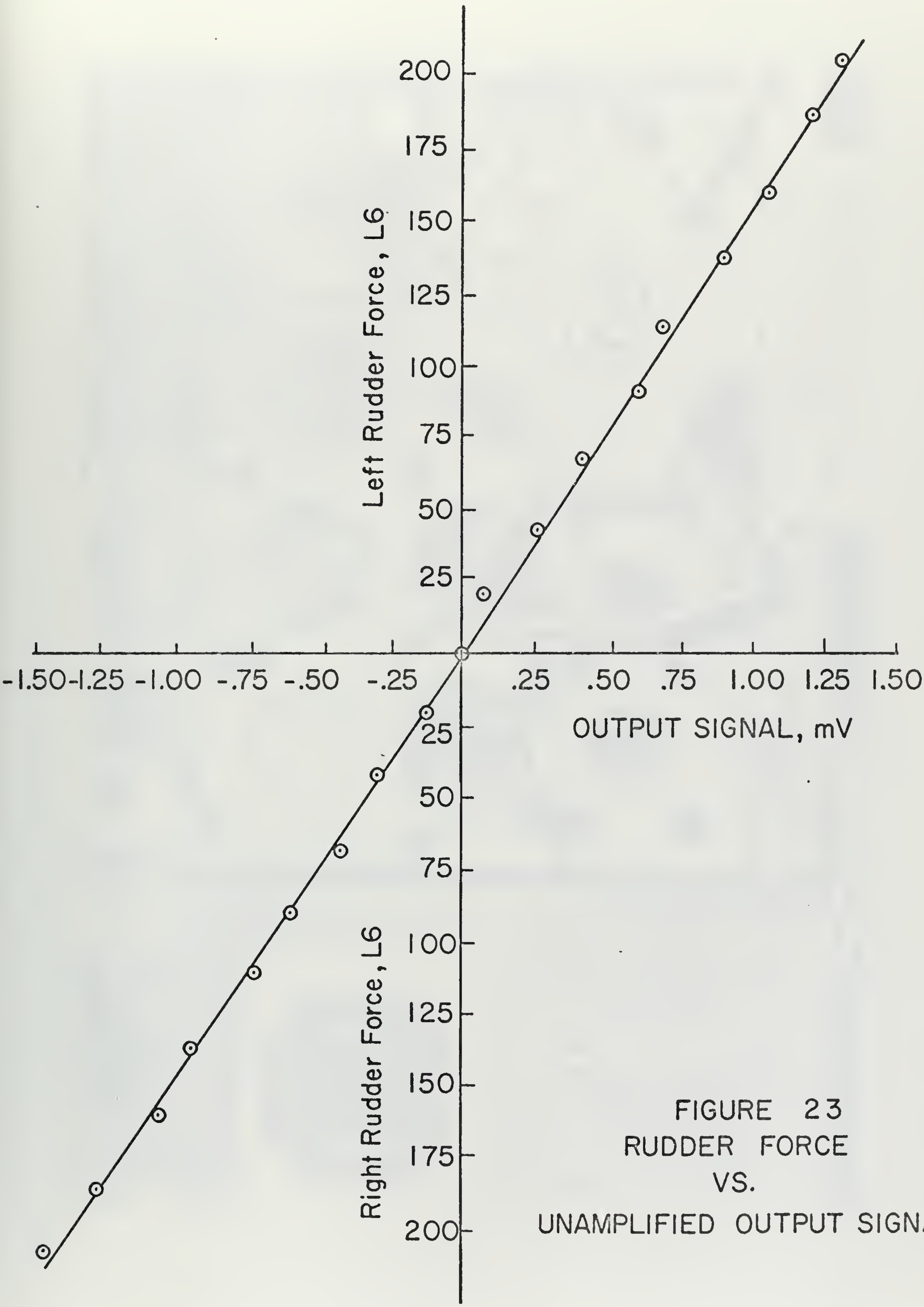


FIGURE 23
RUDDER FORCE
VS.
UNAMPLIFIED OUTPUT SIGNAL

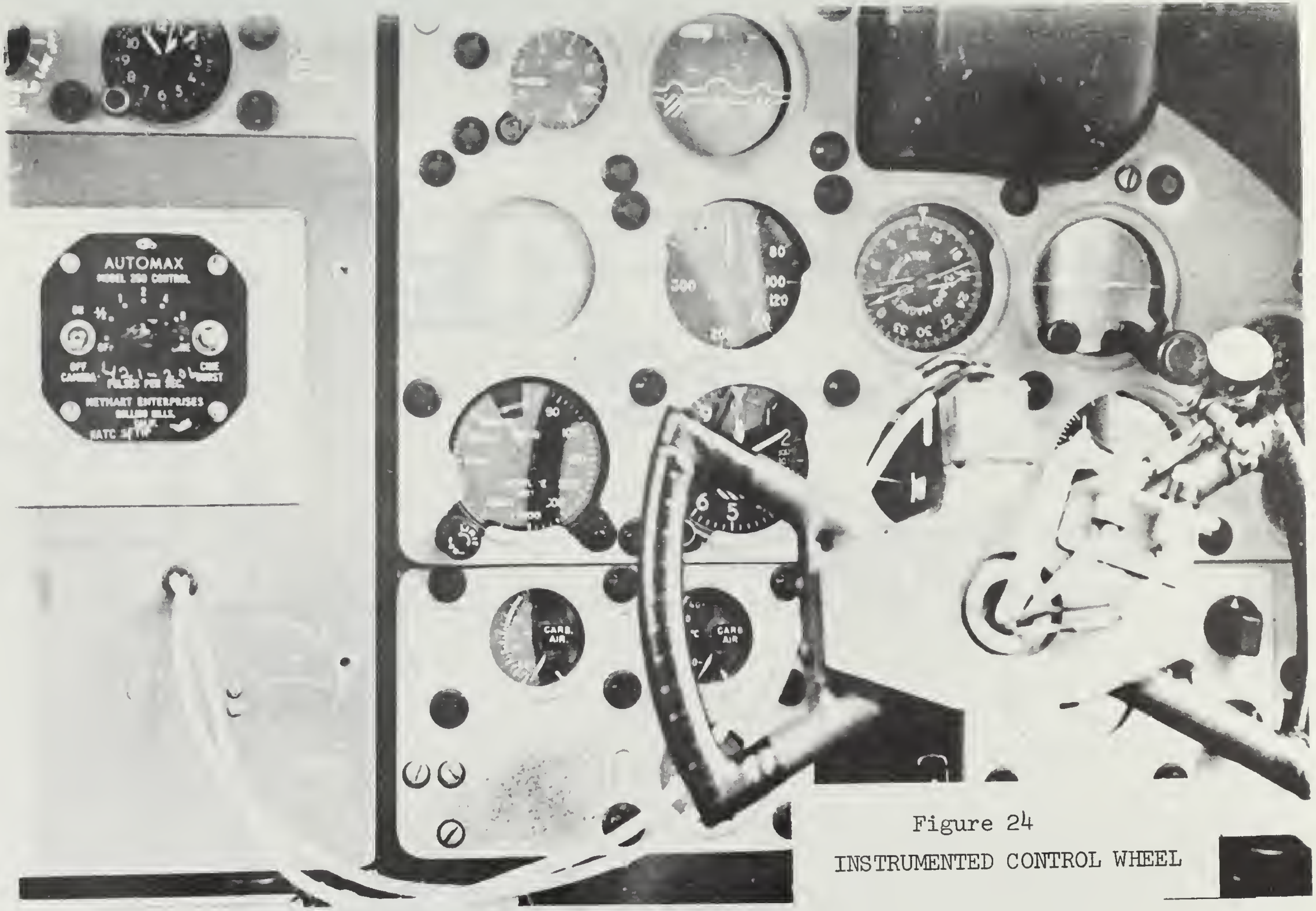


Figure 24
INSTRUMENTED CONTROL WHEEL

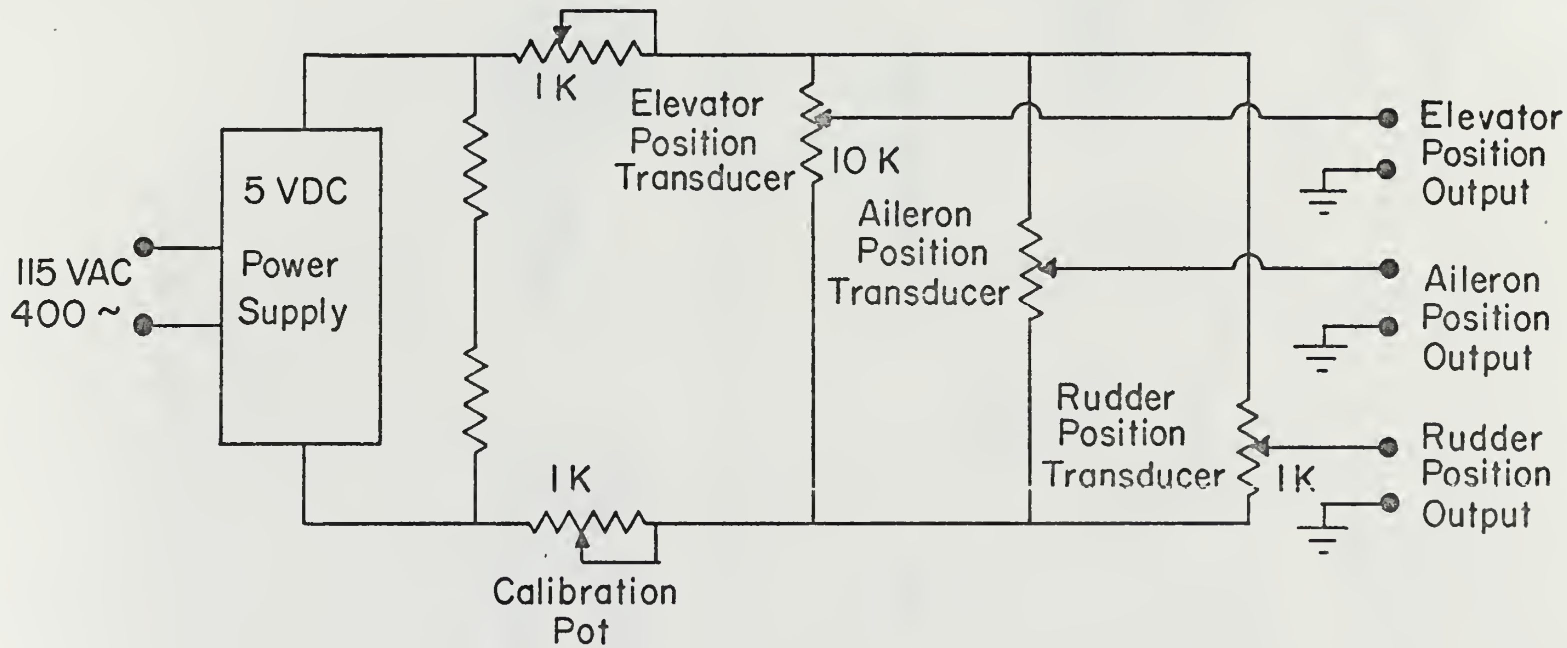
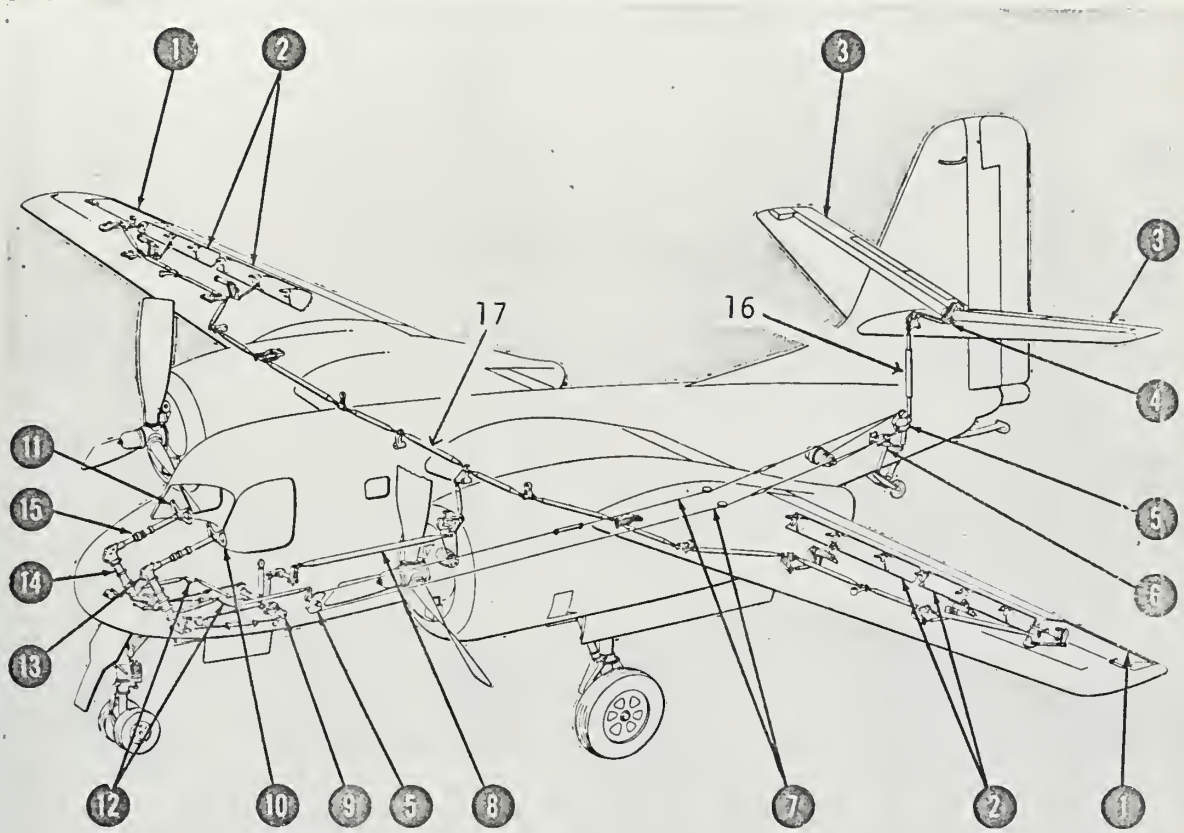


FIGURE 25
DEFLECTION INDICATING SYSTEM

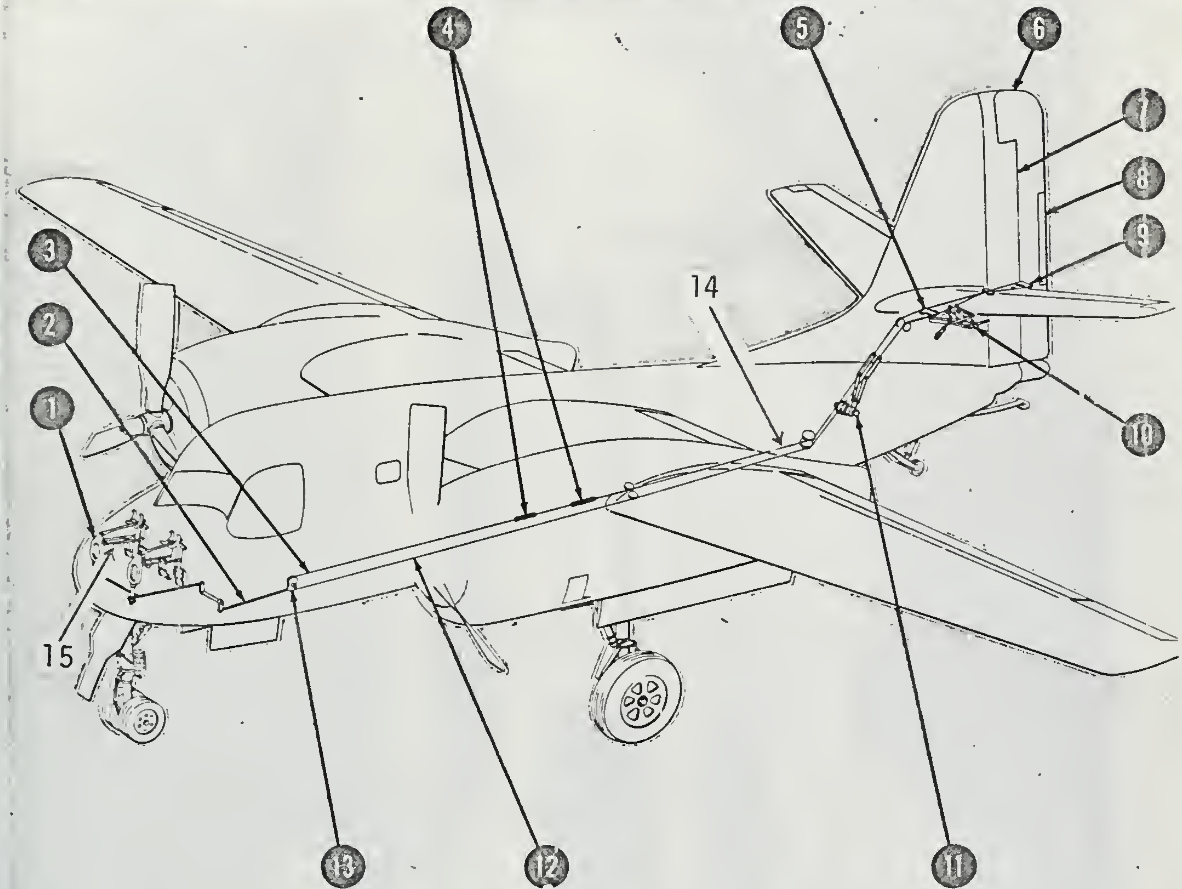


1. AILERON
2. SPOILERS
3. ELEVATOR
4. BELLCRANK ASSEMBLY
5. ELEVATOR CONTROL SECTOR
6. DOWN-SPRING MECHANISM
7. CONTROL CABLE
8. PUSHROD ASSEMBLY
9. SECTOR TUBE ASSEMBLY
10. PILOT'S CONTROL WHEEL
11. CO-PILOT'S CONTROL WHEEL
12. BELLCRANK
13. SPROCKET
14. CONTROL COLUMN
15. CO-PILOT'S CONTROL DISCONNECT
16. ELEVATOR POSITION TRANSDUCER
17. AILERON POSITION TRANSDUCER

Figure 26

AIRCRAFT LONGITUDINAL AND LATERAL

FLIGHT CONTROL SYSTEMS



1. RUDDER PEDAL INSTALLATION
2. PUSH ROD
3. RIGHT RUDDER CABLE
4. FAIRLEAD
5. AFT RUDDER CONTROL SECTOR
6. RUDDER
7. RUDDER TRIMMER
8. RUDDER BALANCE TAB
9. RUDDER HORN
10. RUDDER TRIMMER SYSTEM
11. RUDDER AUTO PILOT SERVO
12. LEFT RUDDER CABLE
13. FORWARD RUDDER CONTROL SECTOR
14. RUDDER POSITION TRANSDUCER
15. RUDDER FORCE TRANSDUCER

Figure 27

AIRCRAFT DIRECTIONAL FLIGHT

CONTROL SYSTEM

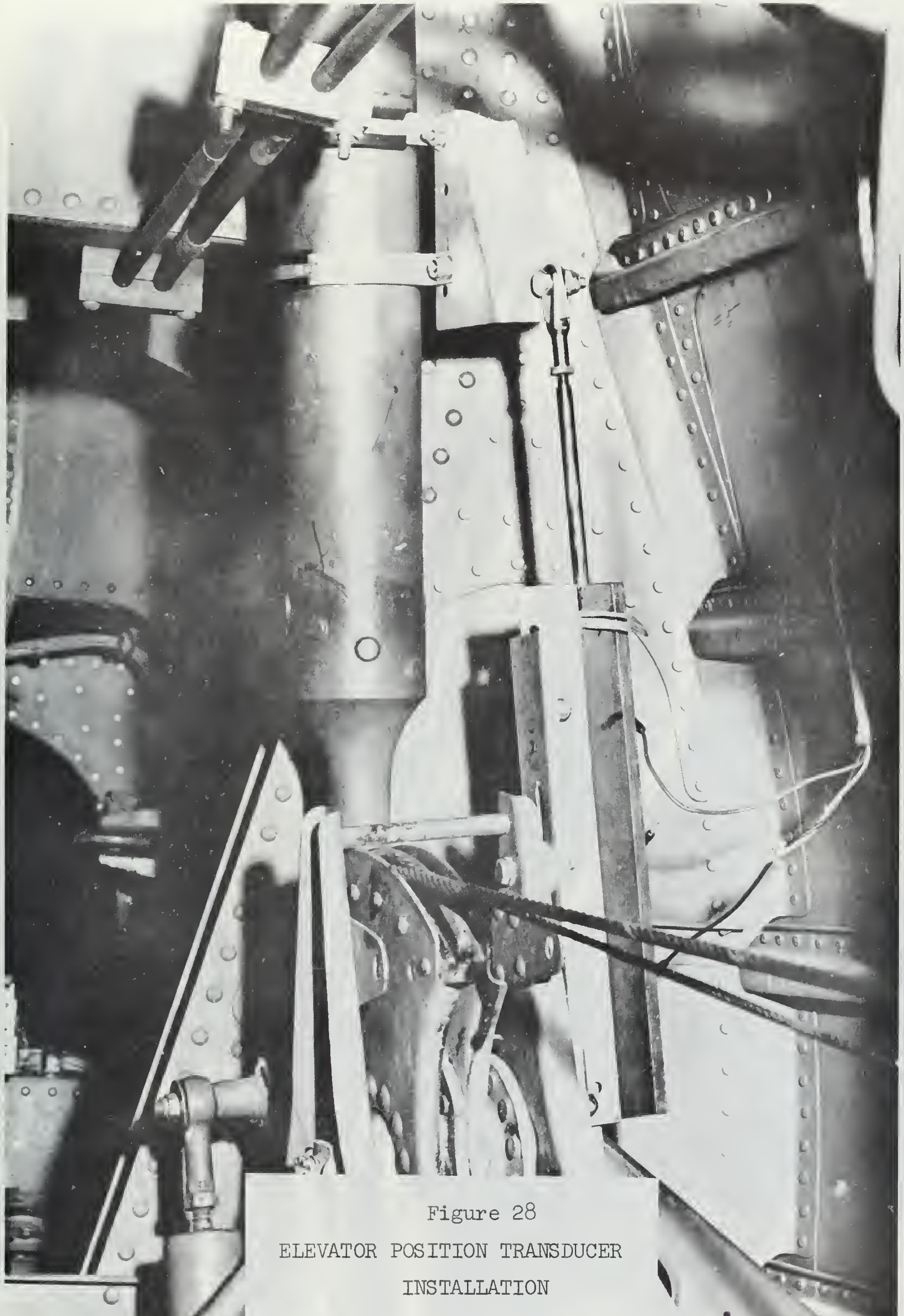


Figure 28
ELEVATOR POSITION TRANSDUCER INSTALLATION

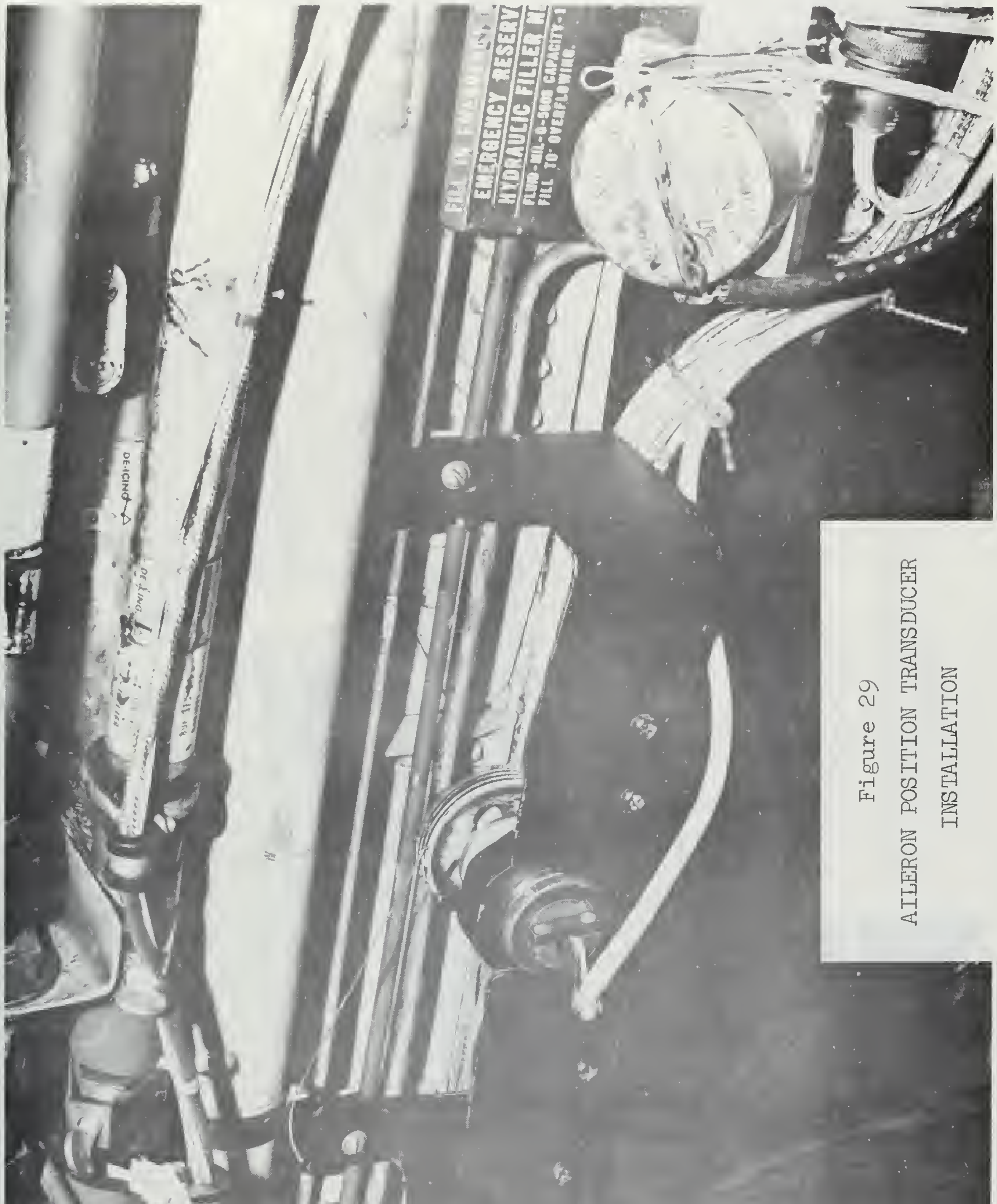


Figure 29
AILERON POSITION TRANSDUCER
INSTALLATION

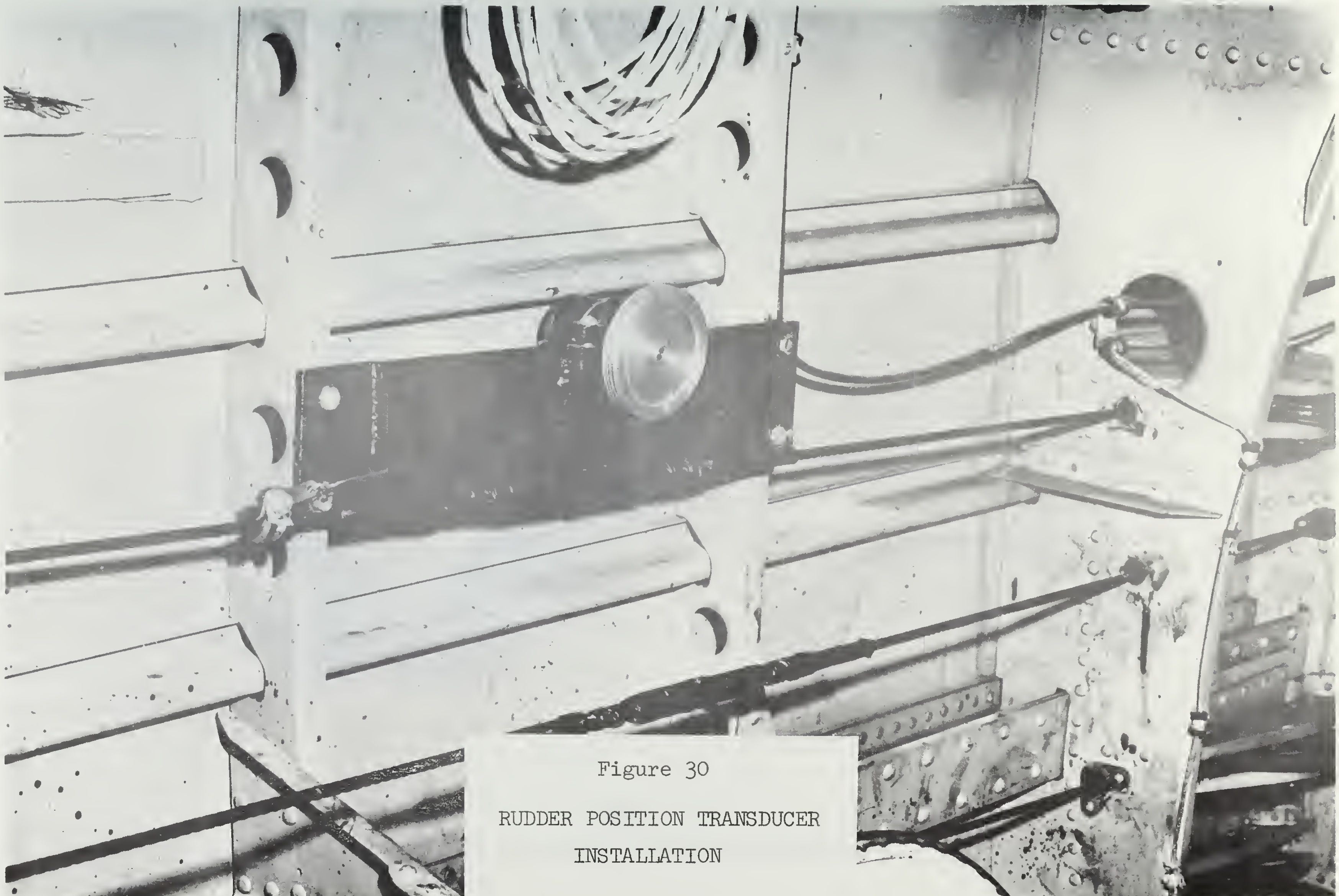


Figure 30

RUDDER POSITION TRANSDUCER INSTALLATION

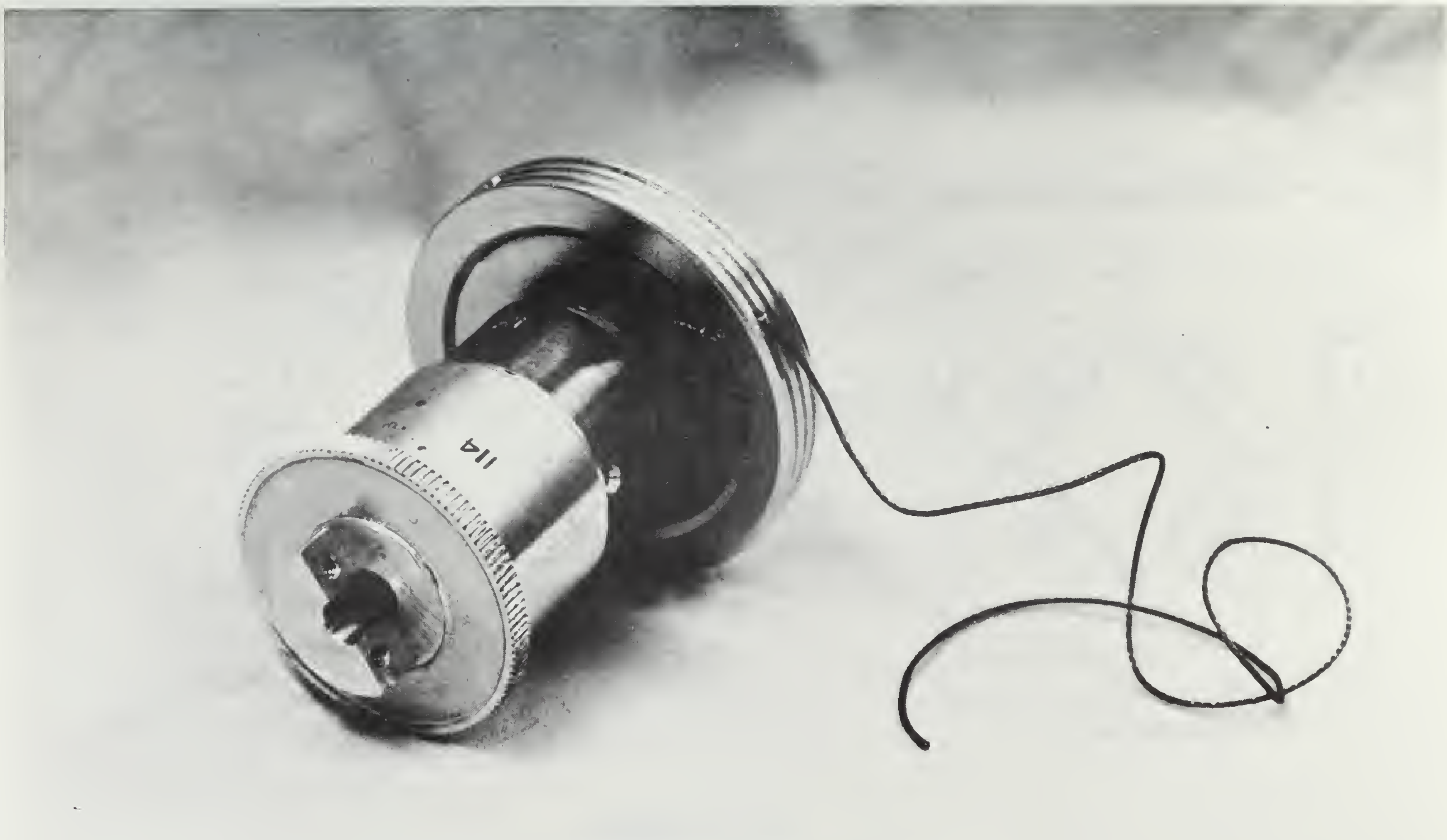


Figure 31

AILERON AND RUDDER POSITION TRANSDUCER

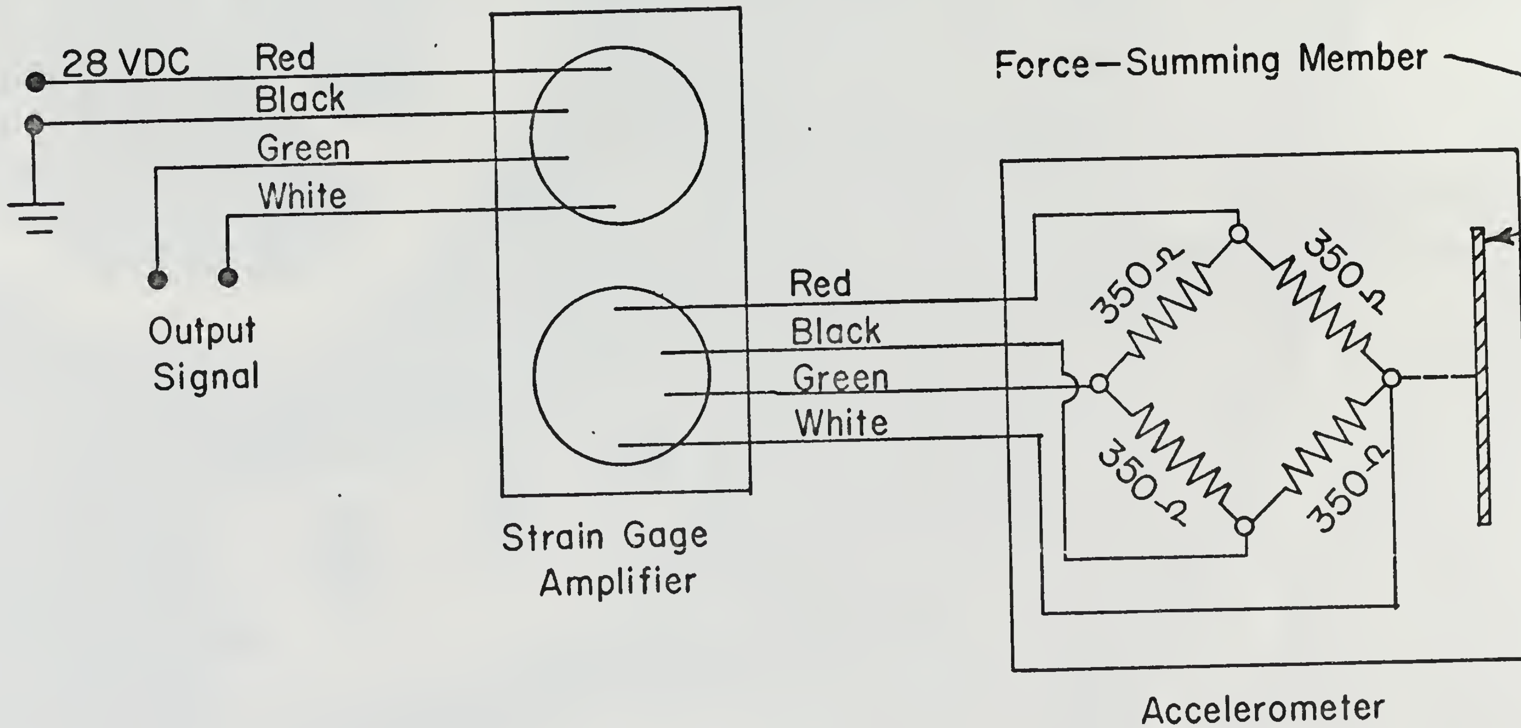


FIGURE 32

NORMAL ACCELERATION SCHEMATIC

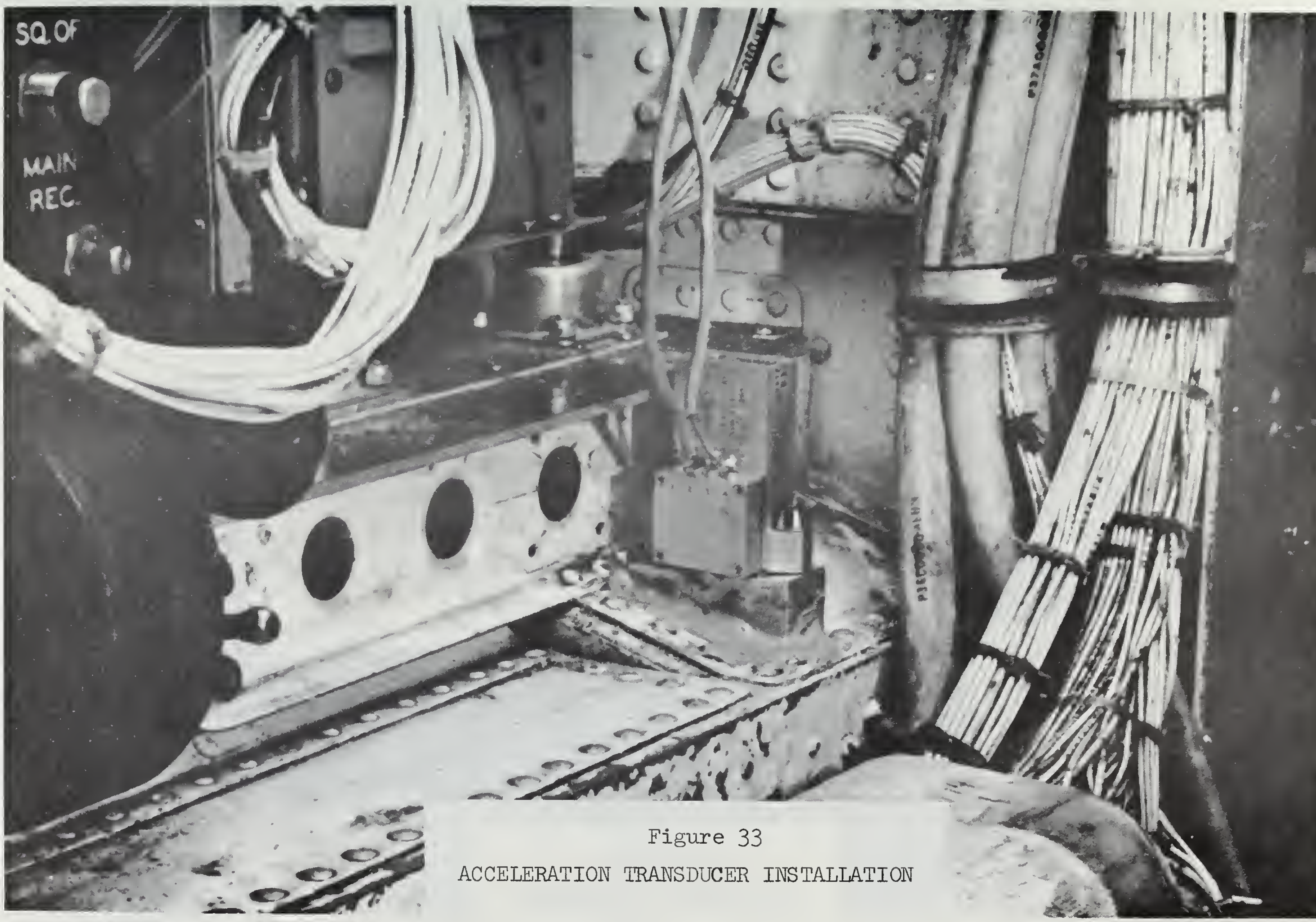


Figure 33

ACCELERATION TRANSDUCER INSTALLATION

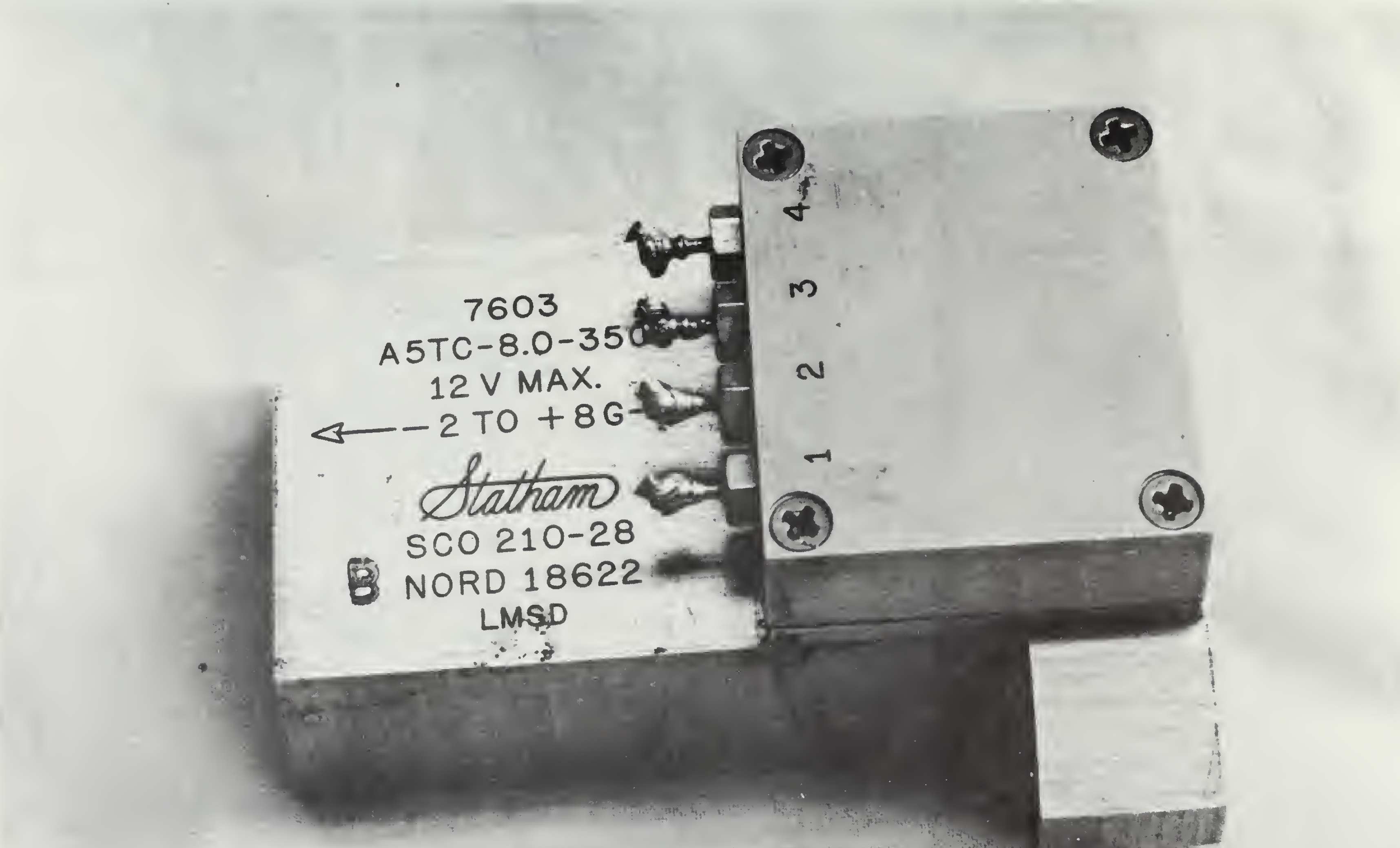


Figure 34
ACCELERATION TRANSDUCER

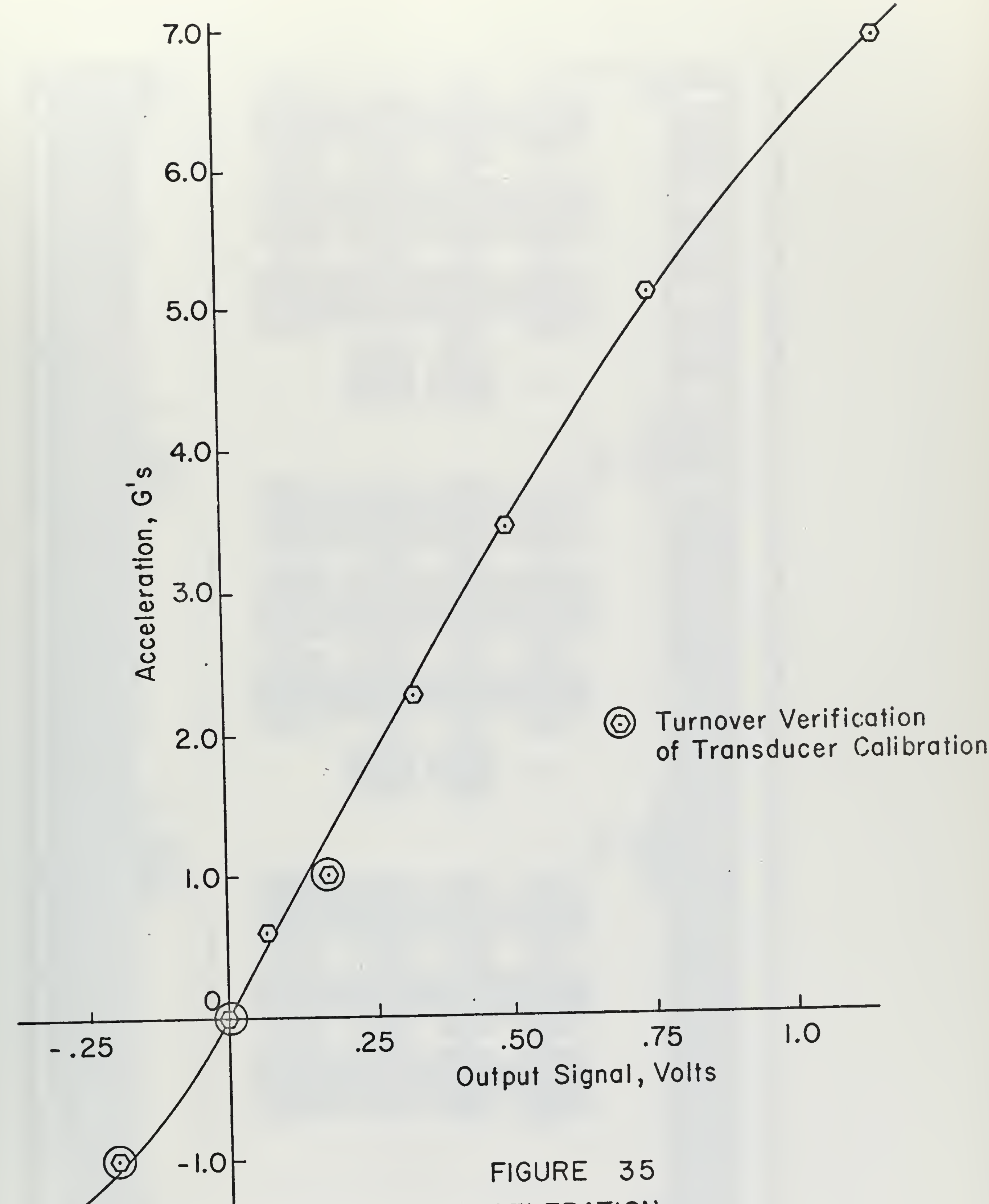


FIGURE 35
ACCELERATION
VS.
OUTPUT SIGNAL

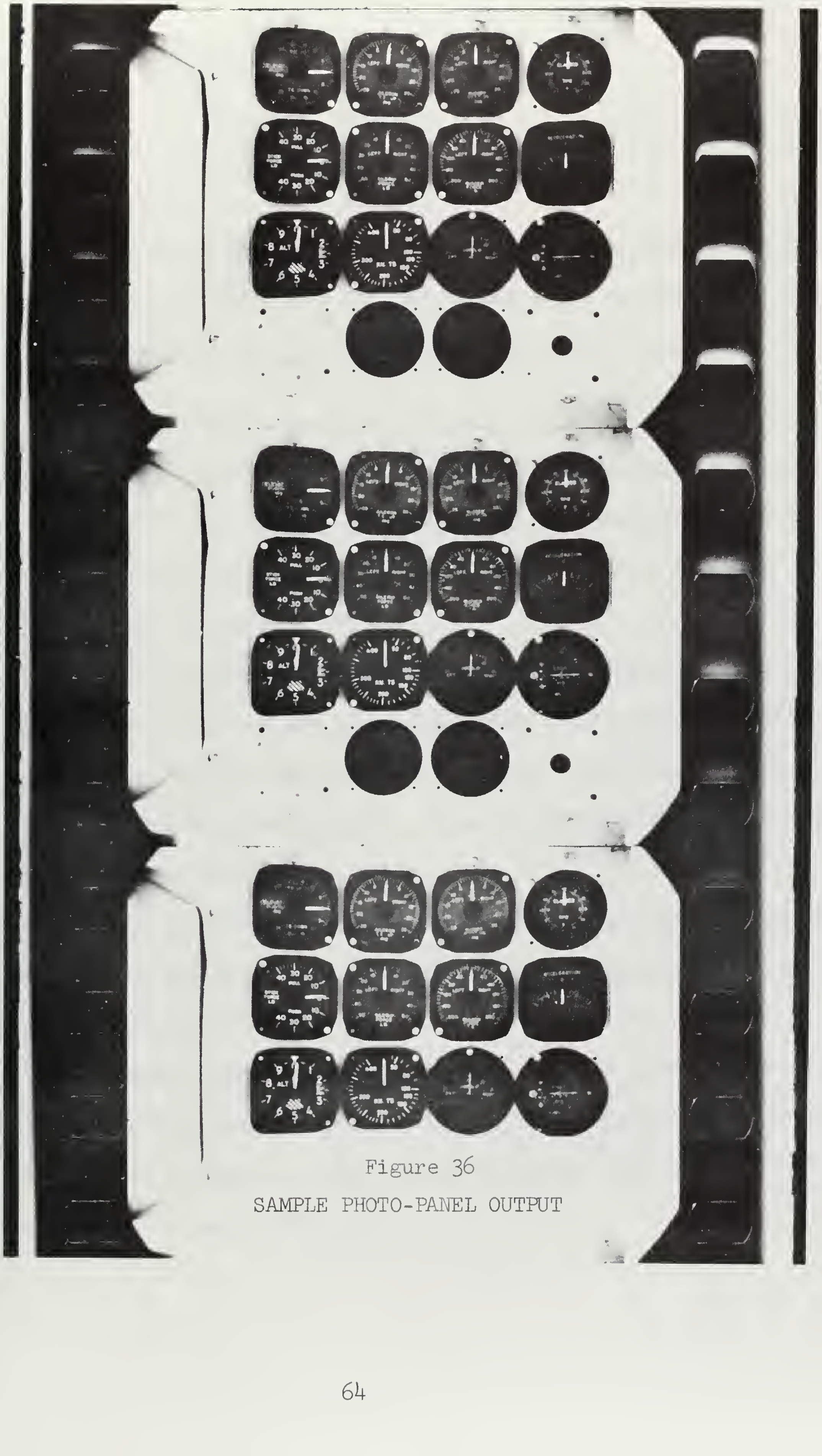


Figure 36

SAMPLE PHOTO-PANEL OUTPUT

APPENDIX B

Calibration Procedures

It is not anticipated that this system will require calibration more than once every three or four months while in use. Only minor changes should be noted in these occasional calibrations.

A. Force amplifier circuit calibration

1) Check power input to balance-box. Bridge power should be $V \approx \pm 15$ VDC. If there is an appreciable inbalance in this ± 15 V, the amplifiers will produce a corresponding error in the plus and minus outputs required.

2) Check bias box for three fresh 1.3 V batteries. Consistantly lower bias voltage is acceptable, however, a low output probably indicates a failing battery whose output will continue to slip. This condition would be intollerable. Bias batteries should be replaced every three or four months.

3) Disconnect bridge inputs to each amplifier and set amplifier output to zero using the balance potentiometer physically located in the amplifier circuit.

4) Before reconnecting the amplifier (or if already connected, with amplifier power secured and bridge power energized) balance each wheatstone bridge using the birdge potentiometer located physically on the top of the balance-box.

5) Leaving the bridges balanced, reconnect them to their respective amplifier circuits and energize same. The physical connections of the bridges across the amplifier input terminal should create a significant amplifier output.

6) Don't zero this output with the wheatstone balance pots. Instead, energize the bias circuit and adjust each bias so that the amplifier outputs are driven to zero.

7) The force indicating system is now calibrated, leaving only the necessity of zeroing the force gages with balance-box (wheatstone bridge) pots prior to each flight.

B. <u>Deflection System Calibration</u>

1) Referring to Figure 25, the two potentiometers mounted on the face plate of the terminal board serve only to balance the positive and negative reference voltages with respect to the phantom ground. Compared to this ground the references should read $V = \pm 1.4$ V. This should be accomplished with the three deflection potentiometers (2 drum, one slide, Figures 28, 29, 30) disconnected.

2) When the panel is balanced, reconnect the deflection pots to the terminal board and physically position them to yield zero output.

3) Connect the potentiometer wire or clamp to its respective aircraft control deflection push rod or cable.

4) Due to the physical "slop" involved in attaching the potentiometer wire or clamp to the control system, numerous attempts may be required to affect the hook-up that yield zero pot output for zero control deflection.

C. Pitch Yaw Calibration

1) Install boom verifying alignment with bracket scribe marks.

2) Install calibration rig (compass rose) under yaw vane and align with zero deflection mark which should be coincident with boom axis.

3) At zero deflection, β should yield zero output. If not, turn vane until allen socket appears in small hole in stem. Loosening this allen socket will free vane from syncro, allowing adjustment.

4) Again check 0° for 0 output. Repeat if necessary.

5) The same technique is required for pitch angle with the exception that zero output is located at five degrees nose-up pitch.

6) If pitch calibration is off, adjust as in the yaw calibration.

7) Note, due to wing station orientation with freestream in flight, $\beta = 0^{\circ}$ in flight might not agree with ground setting which was aligned with the boom. If such an inconsistancy occurs in flight check, re-set vane as in 3) during subsequent ground check.

D. Photo-Panel DC Indicator Calibration

1) With the aforementioned calibrations achieved, the respective DC indicators can be calibrated by adjusting the sensitivity potentiometer, located on the back of each indicator, for a known input of force, deflection, vane pitch, etc.

APPENDIX C

Preflight Procedures

1. Check Battery, Generators, Ignition OFF

2. Install loaded camera on photo-panel

3. Check camera lens settings

    a) Distance 5 feet

    b) f - stop 5.6

4. Place Flight ID card in card holder

5. Boom Installation

    a) Insure scribe marks align

    b) Insure for and aft brackets sit squarely on boom and are tight.

    *c) Check vane guard on with vane points forward

    d) Connect static and total pressure lines matching colors of disconnect fittings.

    e) Connect cannon plug

6. Connect Aircraft to External 28 VDC power.

7. Check red guard switch ON

8. Plug in battery pack

9. Position one person at photo-panel, one in cockpit (both with headsets on).

10. Inverter C.B's in

11. Photo-panel C.B in

12. Turn camera control ON

    a) Check illumination of all lights

    b) Check operation of camera at different frame speeds.

13. Camera OFF

14. Apply aileron, elevator and rudder forces to respective co-pilots controls to check movement of corresponding gages in photo-panel.

15. Gust lock OFF

16. Move all control surface to check movement of corresponding gages in photo-panel.

17. Person in cockpit exit aircraft for vane check

*18. With guard on, move vanes small amount to check movement of gage needles in photo-panel.

19. Secure external power

20. Pull following C.B's

    a) Photo-panel

    b) all inverter (3)

21. Position one person in electronic compartment, one at photo-panel

22. With instructions of person at panel, balance (zero) aileron, elevator and rudder force gages using pots on balance box.

23. Lock pots.

24. Secure red guard switch; leave battery pack plugged in

25. Remove vane guard

26. Secure photo-panel

Inflight Turn-On

1. Turn red guard switch ON

2. Photo-panel C.B in

3. Control camera as desire

Inflight Secure

1. Photo-panel C.B. out

2. Disconnect battery pack (prior to landing)

*3. Secure red guard switch (prior to landing)

Ground Secure

1. Check following:

    a) Camera control OFF

    b) Photo-panel C.B. out

    c) Red guard switch OFF

    d) Battery pack disconnected, plut cover installed.

2. Remove camera from photo-panel

3. Secure panel

*4. Install vane guard, vane points forward

5. Disconnect boom fittings

6. Remove boom

7. Secure boom bracket

8. Install protective covers on pitot and total pressure lines, and cannon plug

*Vane deflections in excess of gage limits overload gages. Ensure RED guard switch is secured any time excess vane deflections might be encountered. e.g. taxi operations.

EQUIPMENT LIST

1. US2A BUNO 136533, Grumman Aerospace Corp.
2. Photo-Panel, NATC Patuxent River
3. Camera, Automax G-1
4. Camera Controller, Automax 250
5. Camera Lens, Agenieux R-11
6. Intervalometer, Automax 25
7. Film Magazine, Triad 700
8. Photo-panel Lights, 28 VDC General Electric Model 307
9. Film, Kodak Linagraph Shellburst, Grey base, 35mm
10. NASA Ames Aircraft Control Wheel
11. Ryan Pitch-Yaw Vane Assembly
12. Fairchild A 727 and A 747 Amplifiers
13. Rudder Force System Strain Gages BLH SR-4, FAB 25-12S 13, 120 ohm
14. Battery Pack: Mallory Duracel 1.35 V Mercury batteries
15. Deflection System Transducers
    a) One 10K-2 Sliding arm transducer
    b) Two 5K-2 Rotary drum transducers
16. Statham Model A5TC-8.0-350 Accelerometer
17. Statham Model CA0-3-12594 Strain Gage Signal Amplifier
18. Stablvolt DC Power Supply Serial No. 5046
19. Altimeter Type MB-1, Serial No. A-44
20. Weston Zero Center a/c Indicators Model 955 Type 61-76 24 2Y1, 1 ma Movement
21. Indicator, Airspeed, Pitot-Static AN5861T2

22) Timer, Elapsed, Part #51A45A1-1 Serial No. 1759

23) Indicator, Zero Center, 100 a Movement

24) Indicator, Zero Center, 25 a Movement

BIBLIOGRAPHY

1. Advisory Group for Aeronautical Research and Development, North Atlantic Treaty Organization, Flight Test Manual, 2d ed., v. 2, 3, 4, Pergamon Press.

2. Bethwaite, C. F., Digital Data Acquisition Systems - Breakthrough or Breakdown, paper presented at Flight Test Instrumentation Internation Symposium, 2nd, England 1962.

3. Bruning, G., Flight Test Instrumentation for V/STOL Aircraft, paper presented at Flight Test Instrumentation International Symposium, 2nd, England 1962.

4. Bureau of Naval Weapons, NAVWEPS 01-855AA-1, NATOPS Flight Manual, Navy Models S-2A-2C-2F Aircraft, 15 January 1966.

5. Department of the Navy, Military Specification MIL-F-8785 (ASG), Flying Qualities of Piloted Aircraft, 1 September 1954.

6. Langdon, S. D., et al., Naval Test Pilot School Flight Test Manual, Naval Air Test Center, Patuxent River, Maryland, August 1966.

7. Neyhart Enterprises, Operating Instructions, Automax Data Recording Cameras.

8. Norton, H. N., Handbook of Transducers in Electronic Measuring Systems, Prentice-Hall, 1969.

9. Perry, M. A., Flight Test Instrumentation for Teaching and Research at the College of Aeronautics, paper presented at Flight Test Instrumentation International Symposium, 1st, England, 1960.

10. Rioland, D. M., Instrumentation Requirements for the Dynamic Testing of Aircraft, paper presented at Flight Test International Symposium, 1st, England, 1960.

11. Townsend, M. W., Naval Test Pilot School Performance Testing Manual, Naval Air Test Center, Patuxent River, Maryland, August 1966.

12. United States Naval Test Pilot School, Guidelines for Flight Reports, November 1969.

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13. ABSTRACT

Various methods for investigating the stability and control characteristics of the US2A were considered in obtaining a system that would provide a proper degree of data accuracy, data availability and system reliability yet still be instructional and functional. To this end, a photo-panel system with its various input systems was designed and incorporated into the Aeronautics Department's US2A, BUNO 136533. Installation and component check-out of this photo-panel system was achieved at the Naval Postgraduate School during the period of July 1970 to February 1971. Stability and control flight evaluation utilizing the system was not completed, however, due to an aircraft accident involving US2A BUNO 136533.

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
photo-panel						
instrumentation						
vane assembly						
synchro						
calibration						
wheatstone bridge						

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